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## THE EFFECT OF ALTITUDE ON BOMBER PERFORMANCE

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MEMORANDUM REPORT

for the

Army Air Forces, Materiel Command

# THE EFFECT OF ALTITUDE ON BOMBER PERFORMANCE

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## INTRODUCTION

A series of reports, references 1 to 4, has been directed toward relating the performance of bombers to their design parameters; namely, power, gross weight, wing area, and altitude. One report, reference 5, shows the effect on performance of variation in the efficiency parameters: power plant, aerodynamic, and structural efficiencies. In all the studies the effect of design and operating altitude on the performance has been largely submerged while attention has been focused on the other parameters. The present study is devoted to the analysis of the effect of design and operating altitude on performance.

The chief emphasis in this report is placed on range performance and charts are first presented giving range as a function of wing loading, power loading, and design altitude. Performance selection charts are then presented which show the high speed, the rate of climb at design altitude, the take-off run at sea level, and the range. The charts are presented for airplanes with design altitudes of 10,000, 20,000, 30,000, and 40,000 feet. These charts consist of performance curves on coordinates of power loading and wing loading so that altitude comparisons may easily be made for constant values of these fundamental parameters.

The selection of the basic data of weights, drags, engine economy, and cooling power were made to correspond to current Air Force practice or to high altitude designs under development for the Air Forces. The values of these factors vary with altitude. To find the variation in performance with altitude requires a careful evaluation of basic data and its variation with altitude. Accordingly, the mathematical representation of basic data and the computations of airplane performance with altitude have been made in greater detail than in previous reports of this series. For example, propellers have been carefully selected and the efficiency computed for all flight conditions, while for cruising flight the engine speed was assumed to be adjusted to give the maximum ratio of propeller efficiency to specific fuel consumption.

This analysis is based on airplanes using four 2000-horsepower engines. The results, however, in general apply to airplanes with other numbers of engines. Weights and wing areas are varied to cover a wide range of power loading and wing loading. In this report, comparison of performance or other characteristics at various altitudes is made at equal power loading (equal gross weight) and equal wing loading (equal wing area). To facilitate such comparisons and as well to make clear the effect of power loading and wing loading on performance at any altitude, performances and other characteristics are presented by means of constant value contours on a coordinate system having power loading as ordinate and wing loading as abscissa.

## ANALYSIS

Range computations are made by the use of the well-known Breguet formula.

$$\text{Range} = 375 \int_{We}^{W} \frac{\eta}{c} \frac{L}{D} \frac{dw}{w}$$

where

$\eta$ propeller efficiency

c specific fuel consumption of the engine, pounds per brake horsepower-hour

L/D lift-drag ratio of the airplane

w airplane weight

W airplane gross weight

We airplane empty weight (gross weight less fuel, oil, and bombs)

In previous reports all range computations have been made for flight at sea level at maximum L/D. In this study range computations are made at maximum L/D and at constant power for various operational and design altitudes. Each factor entering into the range computations is discussed separately.

Range at Maximum L/D

Figure 1(a) shows the specific fuel consumption, pounds per brake horsepower-hour, required to fly the 10,000-foot design altitude airplane at maximum L/D and full gross weight at 10,000 feet altitude. Figures 1(b), 1(c), and 1(d) show the specific fuel consumption required to fly the 20,000-foot, the 30,000-foot, and the 40,000-foot design altitude airplanes for the same conditions at their respective altitudes. Taking a power loading of 13 and a wing loading of 40 to illustrate the effect of altitude on specific fuel consumption at maximum L/D, we note that the specific fuel consumption at 10,000, 20,000, 30,000, and 40,000 feet is, respectively, 0.43, 0.46, 0.52, and 0.66. This approximate 50-percent increase in specific fuel consumption at 40,000 feet over 10,000 feet implies a serious reduction of range at altitude. The increased specific fuel consumption with altitude is the result of the increased power required to operate at maximum L/D at the higher altitude.

Changes in empty weight are caused by changes in the weight of equipment carried by the airplane. Although all of the engines are of the same size and rating, the complete power-plant weight increases with the design altitude. An increase in altitude requires an increase in the weight of the supercharger installation, an increase in weight of the intercoolers and ducts, and an increase in propeller size and weight. Also, for the higher altitudes, cabin supercharging equipment is required. All increases in weight of equipment results in a decrease in disposable load. Figures 2(a), 2(b), 2(c), and 2(d) show the disposable load on coordinates of power loading and wing loading. The disposable load as given here consists of bomb load, fuel, and oil. For a power loading of 13 and a wing loading of 40, for example, the disposable load for design altitudes of 10,000, 20,000, 30,000, and 40,000 feet is $38\frac{1}{2}$, 37, $35\frac{1}{2}$ and 33 percent of the gross weight, respectively. This effect alone would decrease the range of the 40,000-foot design altitude airplane 15 percent with respect to the 10,000-foot design altitude airplane.

The propellers were selected for each airplane from design selection charts (see appendix) for the high-speed condition, and the propeller efficiency was estimated from test data for all flight conditions. It was found that, with careful selection of propellers, losses in cruising efficiency could be held to within 2 or 3 percent of peak efficiency as long as the wing loading was not below about 20 pounds per square foot. If propellers are not carefully selected, a large drop in propeller efficiency for cruising at low power is probable.

The thrust power for cooling was taken as proportional to the brake horsepower. Hence, for performance computations, an allowance for cooling power is expressed as a reduction in propeller efficiency. These reductions are 5, 7, $9\frac{1}{2}$, and 12 percent for operation at 10,000, 20,000, 30,000, and 40,000 feet, respectively.

A change in L/D is effected by a change in parasite drag. The assumed parasite drag of the airplane was made to vary with altitude because of the increased frontal area of the ducts necessary for cooling. Differences in the total drag of airplanes designed for 10,000 and 40,000 feet for a power loading of 13 and a wing loading of 40, for example, are about one percent when flying at maximum L/D. This difference in drag does not include the cooling drag which was charged as a decrease in propeller efficiency as explained above.

The above analysis shows that the important factors affecting the variation of range with design and operating altitude for the maximum L/D conditions of flight are specific fuel consumption, disposable load, and cooling power. The variation of propeller efficiency for optimum propeller designs and the variation in maximum L/D with altitude are relatively unimportant factors.

### Range at Constant Power

The effects of operational altitude on range for operation at constant power are greatly different from the effects for operation at maximum L/D. In the case of operation at maximum L/D an increase in altitude results in a decrease in range. On the other hand for operation at a given power, except for extreme cases of flight at speeds below the speed for maximum L/D, the higher the altitude the greater the average speed. For a constant value of specific fuel consumption, this means that greater range is obtained at the higher altitudes.

The adverse effects of design altitude on the disposable load and parasite drag area was shown in the previous section. These effects tend to reduce range at constant power in exactly the same manner as range at maximum L/D.

For airplanes flying at constant power at design altitude, the above-mentioned effects of operating and design altitude tend to compensate. No definite statement can be made as to the combined effect on range.

## RESULTS AND DISCUSSION

### Range at Maximum L/D

Figures 3(a), 3(b), 3(c), and 3(d) show maximum range of airplanes designed for 10,000, 20,000, 30,000, and 40,000 feet, respectively, and operating at design altitude. Dotted lines give the cruising speed at maximum L/D and gross weight. The shape of the constant range contours for coordinates of power loading and wing loading are similar for the several altitudes. The high ranges appear in the high-power-loading region of the chart at wing loadings of about 30 to 50; low ranges are obtained at low power loading and are not very dependent on wing loading. The most striking difference in the charts is the limitation imposed by the maximum continuous power (1675 hp per engine) at various altitudes on the allowable power and wing loading (dashed lines on the figures). Airplanes defined by power loadings and wing loadings above and to the right of this line have insufficient power to cruise at maximum L/D and gross weight. Hence the range curves are discontinued at this line. The 40,000-foot high-power-loading airplanes are limited to a low-wing-loading region, thus imposing a handicap on range possibilities. Thus, a large gain in range may be obtained by selecting an airplane designed to cruise at low altitude rather than high altitude. For instance, the maximum range with a 10,000-foot airplane is 7750 miles while the maximum at 40,000 feet is only 4500 miles. On the other hand, the 10,000-foot airplane has an initial maximum L/D cruising speed of about 200 miles per hour while the 40,000-foot airplane has an initial cruising speed of about 240 miles per hour.

Figure 4 is a cross plot of maximum L/D ranges given in figure 3. Three airplanes represented by points A, B, and C of figures 3(a), 3(b), 3(c), and 3(d) illustrate the variation of range with altitude for particular values of wing loading and power loading. The differences of ultimate range between 10,000- and 40,000-foot airplanes at design altitudes are of the order of 2000 miles.

Figure 5 is also a cross plot of figure 3 and shows the maximum L/D range at design altitude for the airplanes with a wing loading of 50 pounds per square foot. Each curve is for a particular altitude. Besides showing the change in range with altitude this figure demonstrates the increase in range possibilities from the possible increase of power loading at the lower altitudes.

Figure 6 is a range chart for the 40,000-foot airplane operating at maximum L/D at 10,000 feet altitude. A repetition of curves from figure 1(d) (range at 40,000 feet) is included (dotted line) for easy comparison. This figure shows the range advantage of flying the 40,000-foot airplane at 10,000 feet. For example, at a power loading and wing loading of 13 pounds per horsepower and 40 pounds per square foot, the (maximum) range is increased from 3900 to 5000 miles. This is because the increased horsepower required to fly at 40,000 feet necessitates a higher specific fuel consumption and the power required for cooling is a greater percentage of the brake horsepower.

By means of four sample airplanes, each having a power loading of 13 pounds per horsepower and a wing loading of 40 pounds per square foot, figure 7 shows the range at maximum L/D as a function of operating altitude. It is seen that the range for each airplane decreases with increasing altitude. The airplane designed for 10,000 feet has a greater range at 20,000 feet than the airplane designed for 20,000 feet. A similar circumstance occurs between the 20,000-foot airplane at 30,000 feet and the airplane designed for 30,000 feet. This difference is, of course, due to the greater weight of equipment built into the higher altitude airplanes. However, the advantage in range is obtained at the expense of climb and speed performance.

## Range at Constant Power

Figure 8 shows the range of airplanes operating at maximum continuous power (1675 hp) at design altitude. This is the flight condition in greatest contrast to flight at maximum L/D; this condition giving the shortest range, maximum L/D the longest. In spite of this great difference the range contours bear a striking similarity in the two cases, except for the presence of a somewhat unimportant optimum wing loading for the maximum L/D condition. The increase of range with power loading is marked in both cases. Range at other flight conditions between these extremes is affected by power loading in a similar manner. Comparison of figures 8(a), 8(b), 8(c), and 8(d) shows the high-altitude plane has a slight advantage except at low power loading.

The four curves of figure 9(a) represent four airplanes, each with a power loading of 13 and a wing loading of 40, having design altitudes of 10,000, 20,000, 30,000, and 40,000 feet, respectively, and flying at 10,000 feet altitude. These curves demonstrate the effect of design altitude on range for flight at constant power and cover flight conditions from minimum power at gross weight to maximum continuous cruising power. The difference in range for cruising at a constant cruising power at the same altitude is in favor of the lowest design altitude airplane. This is chiefly due to changes in the weight of equipment with a small effect due to drag coefficient. These effects are general and apply to all the airplanes. In this particular example the propeller of the 40,000-foot airplane is slightly underloaded, also causing a slight drop in range.

Figure 9(b) shows the effect of operating one airplane at constant power at various altitudes or air densities. The airplane has a power loading of 13, a wing loading of 40, and a design altitude of 40,000 feet. The curves show that the higher the airplane flies at a given power the greater the range. This is because airplanes at a higher altitude fly faster on a given power and hence farther. A greater range may be realized, however, with a decrease in altitude because level flight may be maintained with a lower percent of the power.

Figure 9(a) gives the effect of design altitude showing the penalty of high-altitude equipment on range. Figure 9(b) shows the range advantage of operating at high altitude in low-density air at a given horsepower. The combined effects of equipment and density shown in figures 9(a) and 9(b) are shown in figure 9(c). Figure 9(c) shows the curves of range for flight at constant cruising power for airplanes of four design altitudes, each flying at its design altitude. At highest powers some range advantage is shown for the higher altitude airplanes. However, because level flight may be maintained at a lower power at the lower altitude, an increase in range may be obtained at the lower altitudes by taking advantage of flight at low power.

## Performance Charts

Figure 10 is a set of performance charts. Each chart gives the take-off run at sea level, the rate of climb at maximum L/D with full military power, high speed, and the maximum L/D cruising range at design altitude. Comparison of selection charts for several altitudes presents a general picture of the variation of bomber performance characteristics with altitude. Charts of this type are useful in selecting a power loading and wing loading to obtain a desired compromise of performance characteristics.

The following table gives the performance as taken from figure 10 for two airplanes represented by points A and C on the figure. The table shows for the two illustrated points how the design altitude performances of maximum L/D range, high speed, and climb with military power vary with design altitude and how the take-off run at sea level varies with design altitude.

PERFORMANCE AT DESIGN ALTITUDE

| POINT A | | | | |
|---|---|---|---|---|
| W/P 16 W/S 30 | | | | |
| Altitude, ft | 10,000 | 20,000 | 30,000 | 40,000 |
| Max L/D range, miles | 6,650 | 6,000 | 5,300 | 4,300 |
| High speed, miles/hr | 250 | 270 | 290 | 315 |
| Climb, ft/min | 850 | 750 | 550 | 250 |
| Take-off run[1], ft | 2,900 | 2,700 | 2,500 | 2,300 |
| **POINT C** | | | | |
| W/P 8 W/S 70 | | | | |
| Altitude, ft | 10,000 | 20,000 | 30,000 | 40,000 |
| Max L/D range, miles | 3,400 | 2,900 | 2,300 | 1,700 |
| High speed, miles/hr | 340 | 370 | 400 | 430 |
| Climb, ft/min | 1,850 | 1,500 | 1,200 | 700 |
| Take-off run[1], ft | 3,500 | 3,500 | 3,200 | 2,900 |

[1]At sea level.

## CONCLUDING REMARKS

In summarizing the effect of altitude on range performance it was found that:

1. The greatest range is obtained for a low-altitude design operating at low altitude with the airplane flying at the maximum L/D condition (constant angle of attack). The penalty is small for increased operational altitude if the wing loading and power loading are small but becomes important for high wing and power loadings.

2. If the flight is made at a constant power greater than that required for the maximum L/D condition at a given altitude at design gross weight, the range increases with operating altitude until the power condition corresponds to that required for maximum L/D.

3. The range obtained for flight at constant power is always less than that for flight at max L/D.

4. An increase in design altitude of an airplane always decreases the ultimate range due to the increased weight of altitude equipment.

5. A comparison of airplanes operating at various design altitudes for the constant power condition of flight may show an increase in range with altitude due to increased operational speed or a decrease in range due to a decreased fuel load. The comparison shows an increase or decrease in range depending on which effect predominates.

6. In general, if the flight is to be made at maximum cruising power, an increase in both design and operating altitude gives increased range and increased speed.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., October 15, 1943.

## APPENDIX

### Power Plants

The airplanes are each powered by engines capable of developing 2000 horsepower at rated altitude. Power plants rated at 10,000 feet are mechanically supercharged by a single-stage blower, those rated at 20,000 and 30,000 feet by a single-stage turbosupercharger, and those rated at 40,000 feet are supercharged by a two-stage turbo-supercharger. The weight of engines and accessories for various altitudes are given in the section on weights. Accessories include oil coolers and aluminum intercoolers of sufficient size for low power consumption at rated altitude. The curve of minimum specific fuel consumption is given in figure 11(a) together with the corresponding engine speed.

In cases of cruising at minimum specific fuel consumption, if the maximum ratio of propeller efficiency to specific fuel consumption was not obtained, the engine speed was adjusted until this ratio was a maximum. Figure 11(b), giving specific fuel consumption as a function of both engine speed and horsepower, supplies the necessary information.

### Cooling Power

Thrust cooling horsepower is taken as proportional to brake horsepower. This assumption makes it possible to account for cooling losses by an equivalent reduction in propeller efficiency. The following table gives the reduction of propeller efficiency assumed to allow for cooling.

| Altitude (ft) | Thrust cooling power (percent brake hp) (Also equiv. reduction of prop. efficiency) | Approximate reduction of brake hp (percent) |
|---|---|---|
| 10,000 | 5.0 | 6.0 |
| 20,000 | 7.0 | 8.0 |
| 30,000 | 9.5 | 11.0 |
| 40,000 | 12.0 | 14.0 |

## Propellers

Four-blade propellers were used throughout this study so that propeller weights were kept uniform.

The propeller efficiency for the range condition was carefully investigated and propellers were selected that were suited for good range performance, with the high-speed condition given less consideration. This study was necessary because, with the engine operating at the speed for minimum specific fuel consumption with the airplane flying at maximum L/D at its design altitude, the propeller may stall and a serious loss in efficiency occur. To avoid the selection of extremely large propellers, propellers were selected to operate with a tip speed of 1.05 times the speed of sound for the high-speed condition.

At a value of V/nD below two, analysis of the experimental data in reference 6 shows that the $C_L$ at the 0.7 radius for peak efficiency varies approximately as the curve in figure 12. Above a V/nD of two, the $C_L$ at the 0.7 radius rises slowly but because of compressibility limitations it was held constant at 0.51. On the same figure the curve of $(\sigma C_L)_{0.7R}$ is shown, $\sigma_{0.7R}$ being held constant at 0.138 for a four-blade propeller. The propeller selection chart (fig. 13) was prepared for propellers having optimum loading distribution but may be used for any efficient propeller. This chart was made from the data in reference 7. Having the forward velocity and tip speed given, the V/nD is computed; the value of $(C_L)_{0.7R}$ is read from figure 12 and the value of $1/\sqrt{P_c}$ read from figure 13. Then the diameter is given by $D = \dfrac{1/\sqrt{P_c}}{\sqrt{\pi\rho\, V^3/2P}}$. For extremely low-speed airplanes the propeller diameters were limited to 19 feet even though the selection chart shows a larger diameter propeller required for highest efficiency. After the diameter and gear ratio were established, the propeller efficiency was determined from the test data of reference 6. The take-off criterion of all propellers was investigated to ascertain that the propellers were unstalled in the take-off range.

Propeller weights were taken from figure 14 in accordance with the diameter.

## Drag

The drag coefficients used are representative of those obtained on modern high-performance airplanes. The wing profile-drag coefficient is taken as 0.0090. The tail drag coefficient based on wing area is taken as 0.0030. The drag coefficient of fuselage and nacelles based on effective frontal area is taken as 0.120. These coefficients combine to give an expression for profile-drag coefficient.

$$C_{D_o} = 0.0120 + 0.12\ F/S$$

where $F$ is the effective frontal area of fuselage and nacelles, and $S$ is the wing area. In addition $C_{D_o}$ is varied with Mach number in the manner shown in figure 15. This variation is in accordance with the assumption that the airplane does not reach the critical Mach number. At any given altitude $F$ is taken to be constant. This allows for the nacelles becoming effectively more submerged in the wing as the gross weight increases. The nacelle frontal areas are increased with altitude to admit increased quantities of cooling air. Estimates of the size necessary are based on maintaining the ratio of entrance velocity to flight speed within a reasonable range for all flight conditions. The resulting values of effective fuselage and nacelle frontal areas are as follows:

| Design altitude (ft) | Effective fuselage and nacelle frontal areas (sq ft) |
|---|---|
| 10,000 | 137.8 |
| 20,000 | 139.2 |
| 30,000 | 141.2 |
| 40,000 | 144.9 |

## Span Factor

An addition to the minimum parasite and ideal induced drag is assumed and expressed as an increase in the induced drag. Thus, the induced drag is divided by a span factor as in the equation

$$D = C_{D_o} qS + \frac{\left(\frac{W}{b}\right)^2}{2\pi q}$$

The value of $e$ is taken as 0.80 in this analysis.

## Aspect Ratio

An aspect ratio of 10 has been used throughout this analysis. The effect of aspect ratio on maximum range and speed is not critical over a wide range of aspect ratio.

## Load Factor

A design load factor of 4 has been used in determining the wing weight. The design loading condition is a bomb load in the fuselage equal to 5 percent of the gross weight and the fuel load distributed in the wings.

## Wing Thickness

A 20-percent wing-thickness ratio at the root chord was used for all airplanes. This wing is thick enough to keep the wing weight reasonable but not thick enough to cause a high drag.

## Weights

From airplane weight studies the following weights were selected:

1. The landing gear is $7\frac{1}{2}$ percent of the gross weight.

2. The fuselage weight varies with the 2/3 power of the gross weight as in figure 16. This makes the weight vary roughly with the surface area.

3. The weight of each engine with accessories is:

| Altitude | Engine weight | Inter-cooler and ducts | Super-charger instal-lation | Controls and starting | Oil cooler and miscella-neous | Total except pro-peller |
|---|---|---|---|---|---|---|
| 10,000 | 2365 | 0 | (a) | 100 | 250 | 2715 |
| 20,000 | 2265 | 230 | 370 | 100 | 250 | 3115 |
| 30,000 | 2265 | 255 | 370 | 100 | 250 | 3240 |
| 40,000 | 2265 | 295 | 555 | 100 | 250 | 3465 |

(a)Weight of supercharger included in engine weight.

4. The assumed weight of cabin furnishings and of armament and armor are given in figure 17(a).

5. The weight of electrical equipment, surface controls, and hydraulic system are given in figure 17(b).

6. The weight of cabin supercharging equipment incorporated in airplane with 30,000 and 40,000 feet design altitude is given in figure 17(c) and is in addition to the item of cabin furnishings.

7. The crew is assumed to vary from six members at a gross weight of 40,000 pounds to nine members at a gross weight of 265,000 pounds. A weight of 200 pounds is allowed for each crew member. An additional 15 pounds of oxygen equipment is installed for each man.

8. Certain weights have a fixed value:

| | |
|---|---|
| Instruments | 400 lb |
| Communications | 600 lb |
| Automatic pilot | 250 lb |

9. Wing weights.

If it is assumed that the weight of a wing is proportional to the amount of metal required to resist the applied bending moments, the following relationship between wing weight and other airplane parameters may be derived:

$$K = \frac{(W - C_1W_2 - W_1)fR^{3/2}S^{1/2}}{W_1t}$$

in which by trial on modern American bombers and pursuit planes the average value of K is in the neighborhood of 100,000. Keeping the same general arrangement of the equation except for the introduction of a taper ratio T and trying all promising combinations of exponents on a series of airplanes yields the following relationship for the least deviation of K from an average value:

$$K = \frac{\left(W - C_1W_2 - W_1\right)^{0.7} f^{0.7} R^{1.4} S^{0.6}}{W_1 t^{0.7} T^{0.1}}$$

The value of K = 2400 was derived from a number of Army Air Force airplanes and is used in the determination of wing weight for this report. The value $C_1$ is taken as 0.85.

10. The weight of tail surfaces is taken as 10 percent of the wing weight.

11. The weight of fuel system is 0.65 pound per gallon of gasoline.

12. The oil system weighs 1.25 pounds per gallon. Sufficient tankage weight is included to obtain maximum range with no bomb load. The tanks are assumed to be carried in the wings.

## Calculation of Performance

Calculations of speed, range, climb, and take-off were all made in conventional manner. High speed is computed at military power (2000 hp per engine). Range is computed by an integration of the Breguet formula. Climb is computed at maximum L/D and full military horsepower. Take-off run is computed for sea level by Diehl's formula (reference 8) assuming a take-off lift coefficient of 1.2.

## List of Symbols

| | |
|---|---|
| $b$ | wing span |
| $C_L$ | lift coefficient |
| $C_{D_o}$ | profile-drag coefficient |
| $C_1$ | coefficient multiplying the distributed load to give the effective distributed load |
| $D$ | airplane drag, except propeller diameter in propeller characteristics |
| $e$ | span factor |
| $F$ | effective fuselage and nacelle frontal area |
| $f$ | design load factor |
| $K$ | wing weight coefficient |
| $L$ | airplane lift |
| $L/D$ | lift-drag ratio |
| $P$ | engine power |
| $P_c$ | propeller power coefficient, $P_c = \frac{P}{\frac{\pi}{4}D^2qV}$ |
| $q$ | dynamic pressure, $q = \frac{1}{2}\rho V^2$ |
| $R$ | aspect ratio; as subscript, propeller radius |
| $S$ | wing area |
| $T$ | wing taper ratio |
| $t$ | wing-root thickness ratio |
| $V$ | airplane velocity |
| $W$ | airplane gross weight |
| $W/S$ | wing loading, pounds per square foot |

$W/P$ power loading, pounds per horsepower - take-off

$W_1$ wing weight

$W_2$ distributed load in wing

$\rho$ mass density of air

$\sigma_{0.7R}$ propeller solidity at 0.7 radius; the ratio of the total blade chord to the circumference

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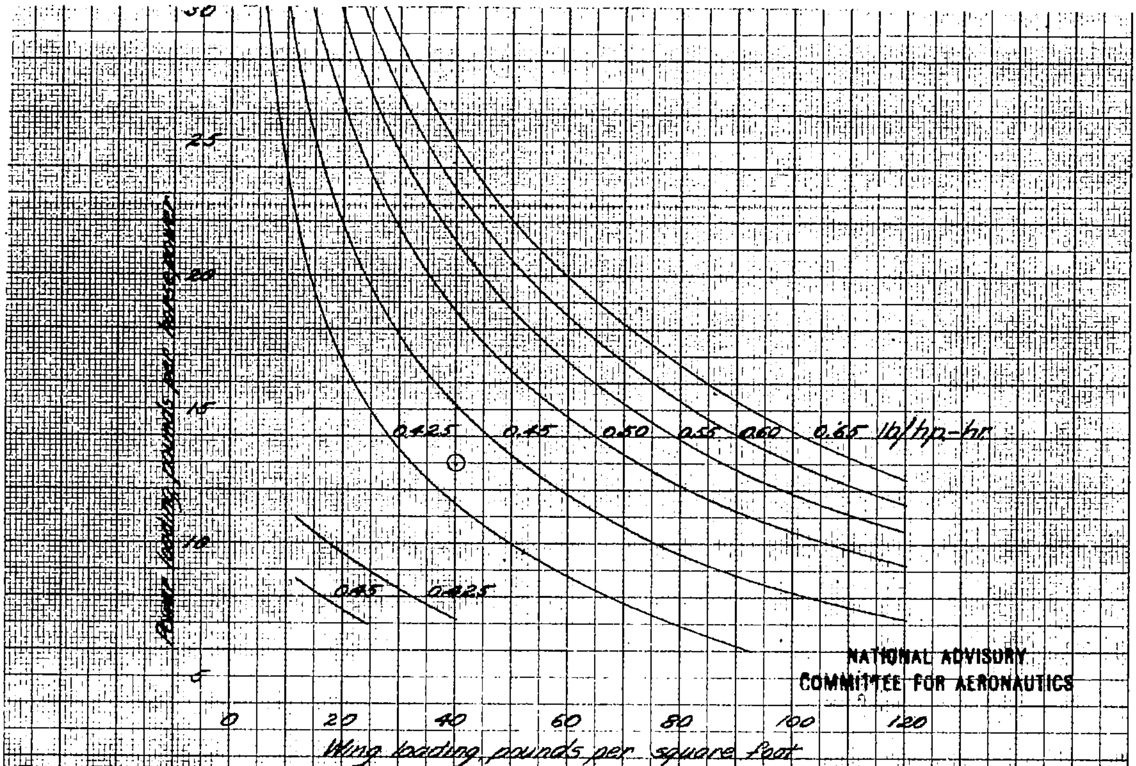



Figure 1(a)- SPECIFIC FUEL CONSUMPTION FOR 10000 FOOT DESIGN ALTITUDE AIRPLANES OPERATING AT MAX. L/D AND DESIGN ALTITUDE.

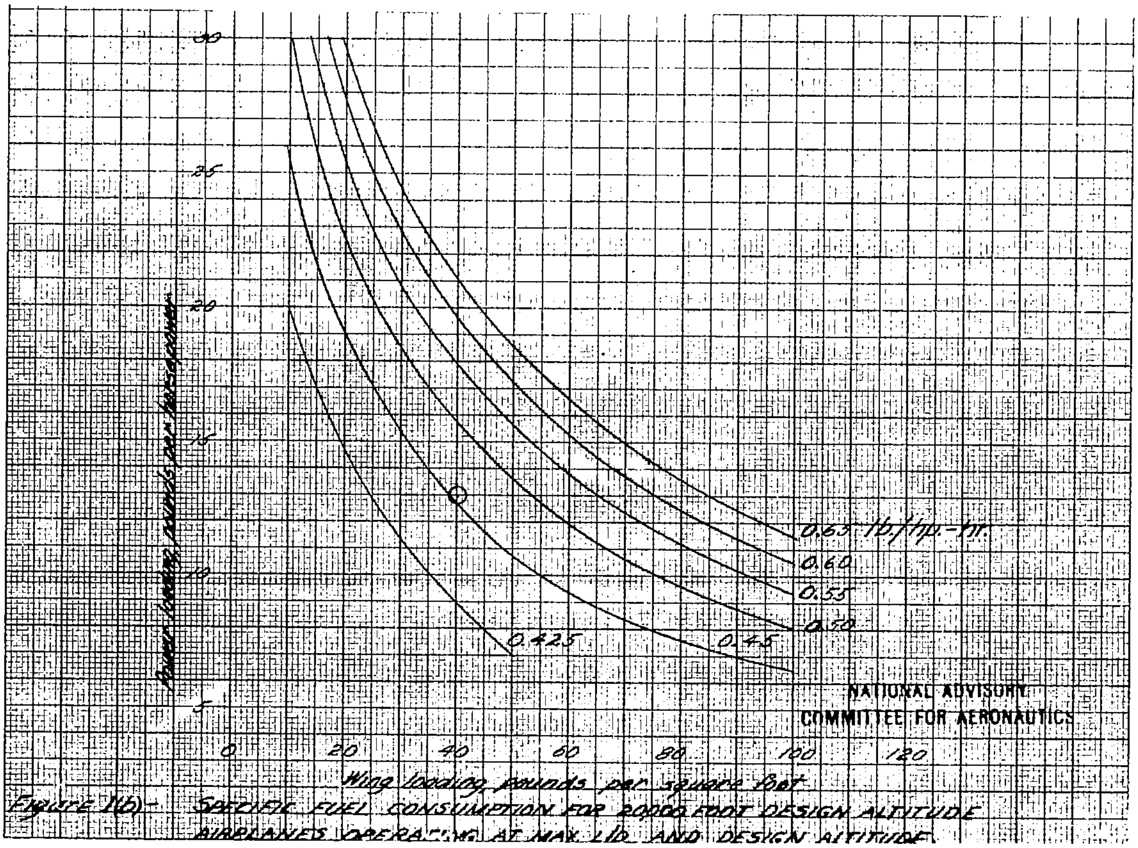



Figure 1(b)- SPECIFIC FUEL CONSUMPTION FOR 20000 FOOT DESIGN ALTITUDE AIRPLANES OPERATING AT MAX. L/D AND DESIGN ALTITUDE.

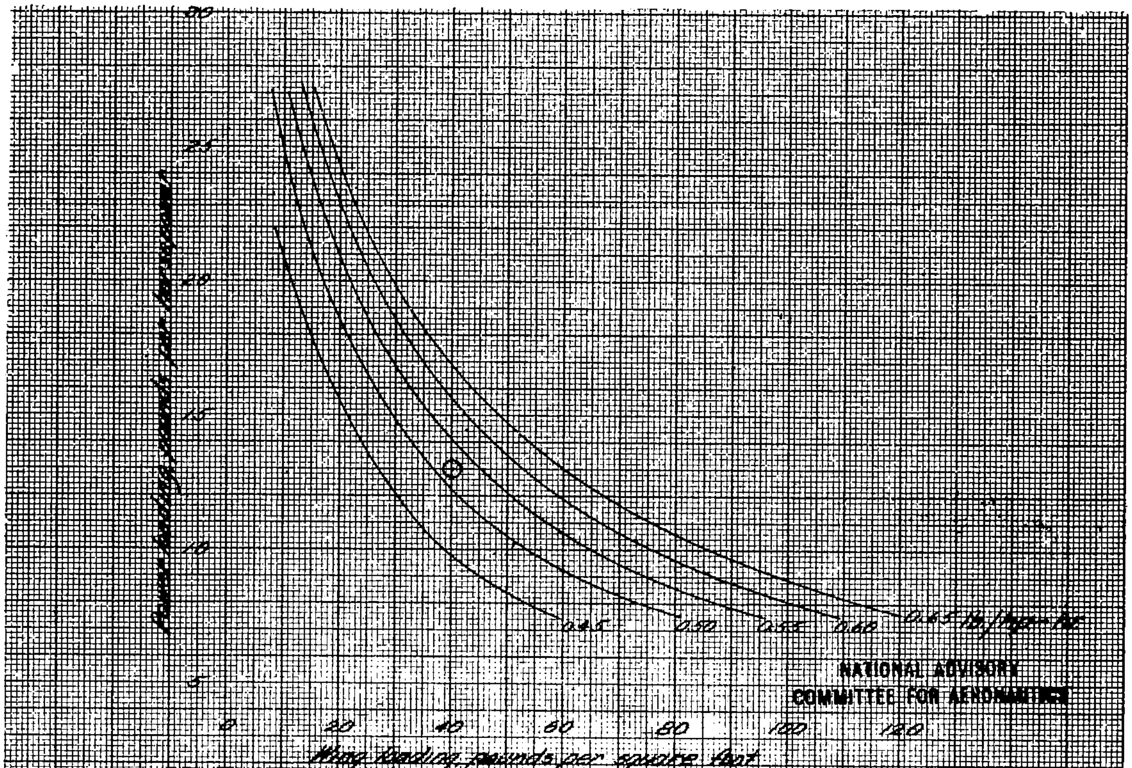



Figure 1(e).- SPECIFIC FUEL CONSUMPTION FOR 30000 FOOT DESIGN ALTITUDE AIRPLANES OPERATING AT MAX L/D AND DESIGN ALTITUDE.

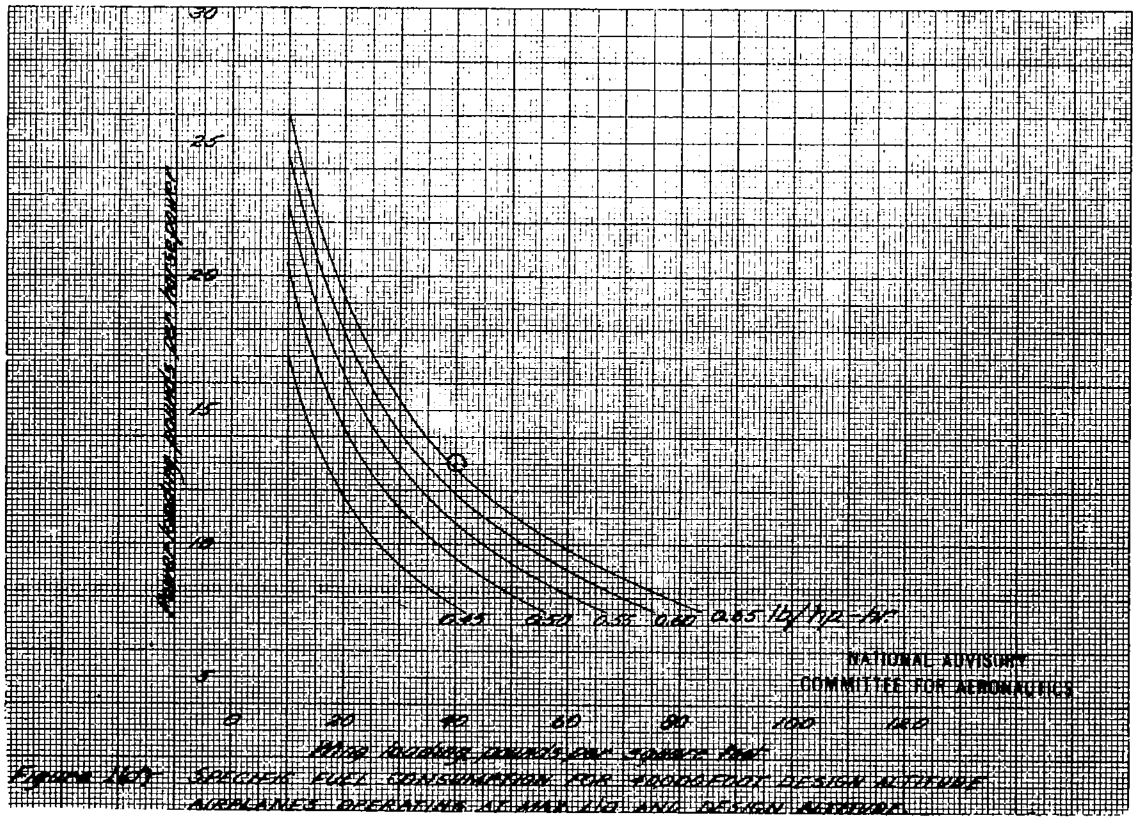



Figure 1(f).- SPECIFIC FUEL CONSUMPTION FOR 40000 FOOT DESIGN ALTITUDE AIRPLANES OPERATING AT MAX L/D AND DESIGN ALTITUDE.

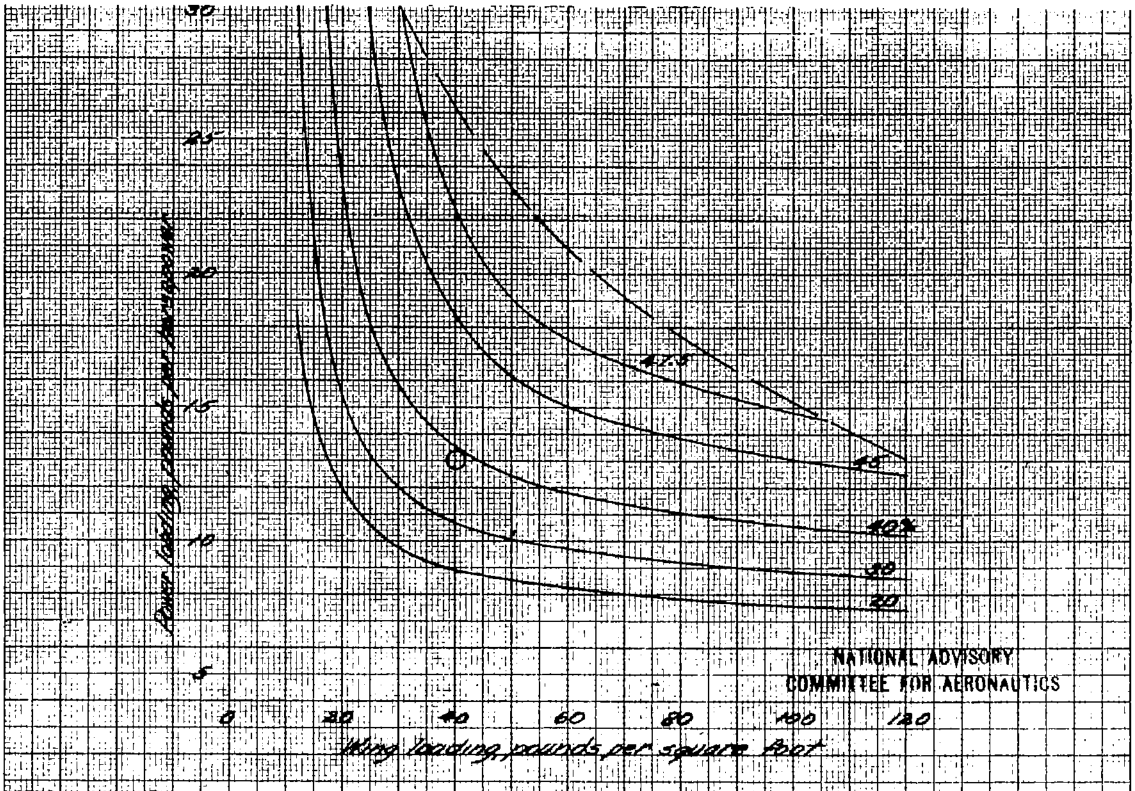



Figure 20. DISPOSABLE LOAD OF 10000 FOOT DESIGN ALTITUDE AIRPLANES, PERCENT OF GROSS WEIGHT

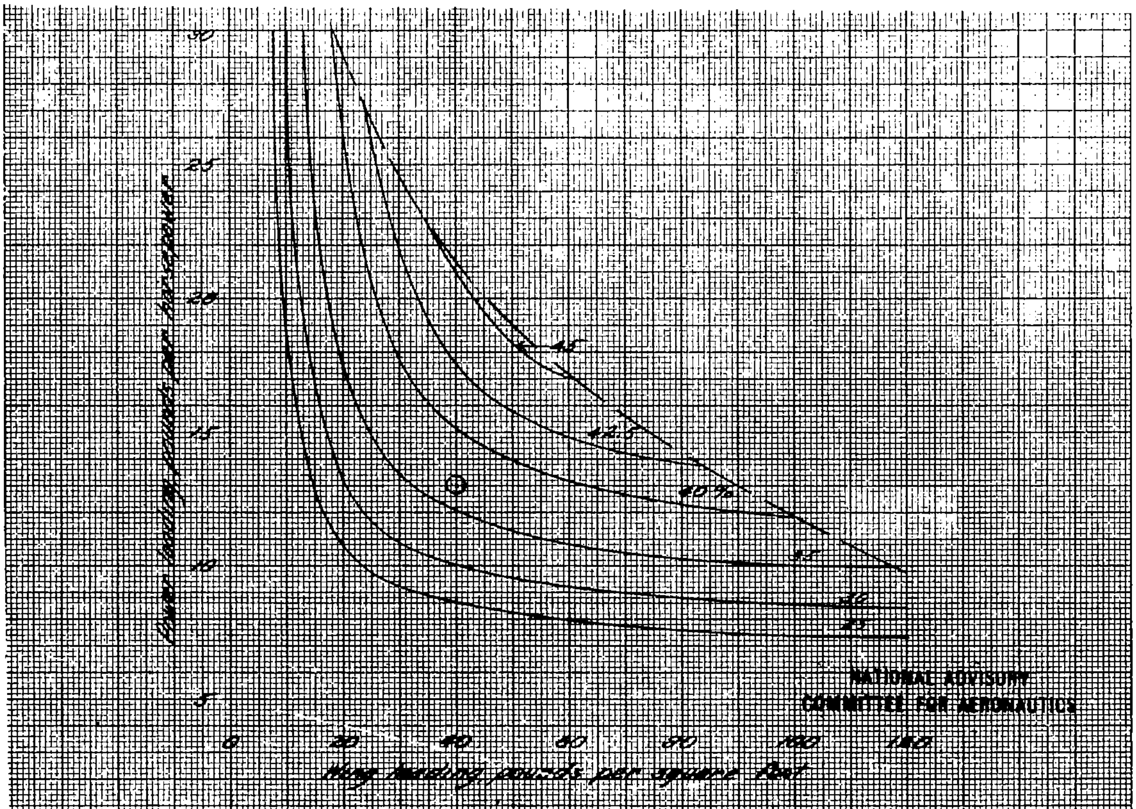



Figure 21. DISPOSABLE LOAD OF 20000 FOOT DESIGN ALTITUDE AIRPLANES, PERCENT OF GROSS WEIGHT

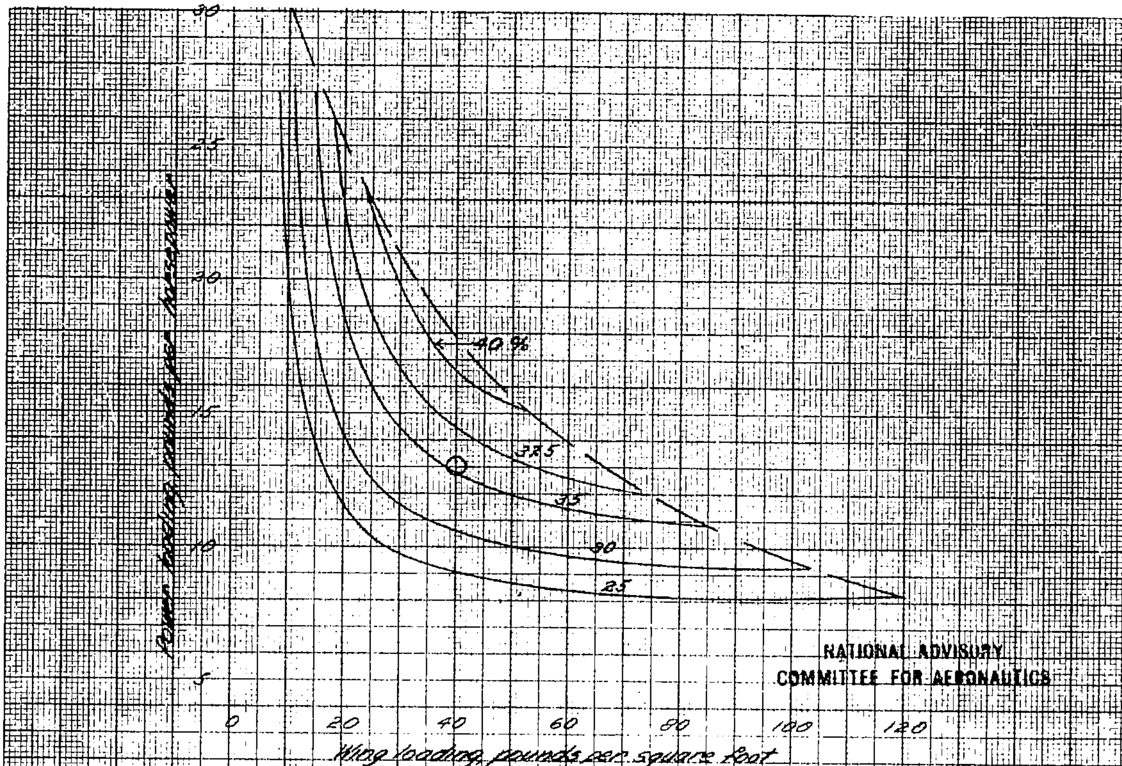



Figure 21(c) – DISPOSABLE LOAD OF 30,000 FOOT DESIGN ALTITUDE AIRPLANES, PERCENT OF GROSS WEIGHT

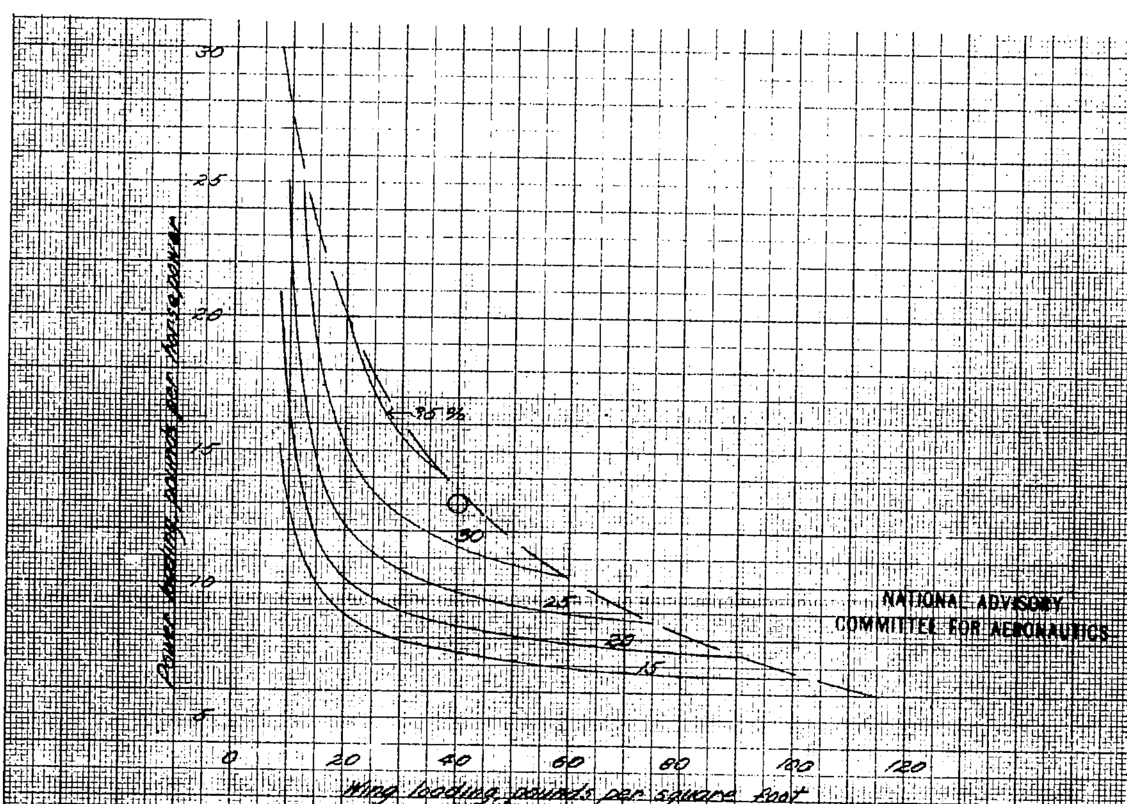



Figure [illegible] – DISPOSABLE LOAD OF 40,000 FOOT DESIGN ALTITUDE AIRPLANES, PERCENT OF GROSS WEIGHT

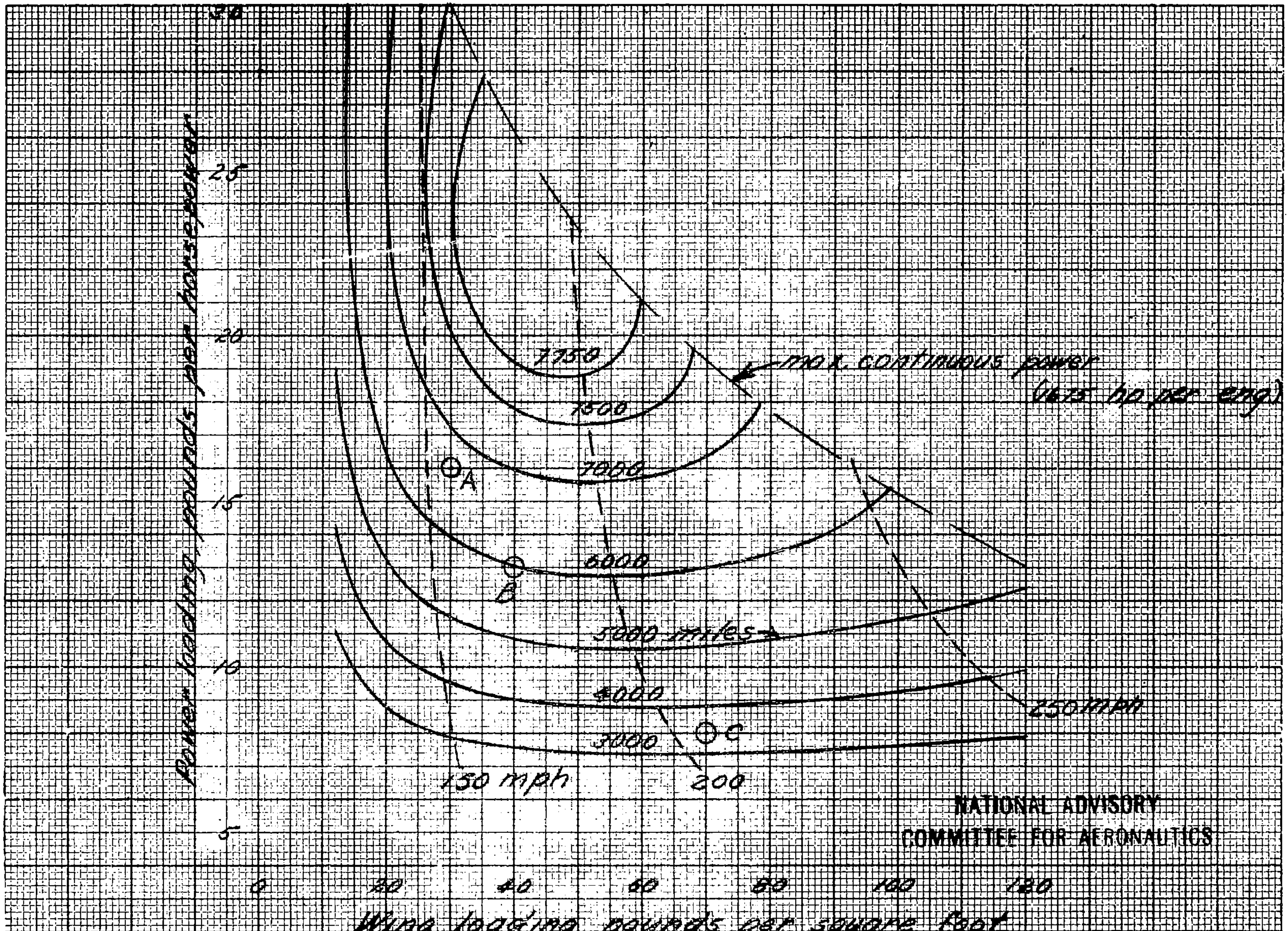



Figure 3(a)- RANGE OF 10,000 FOOT DESIGN ALTITUDE AIRPLANES OPERATING AT MAX L/D AND DESIGN ALTITUDE

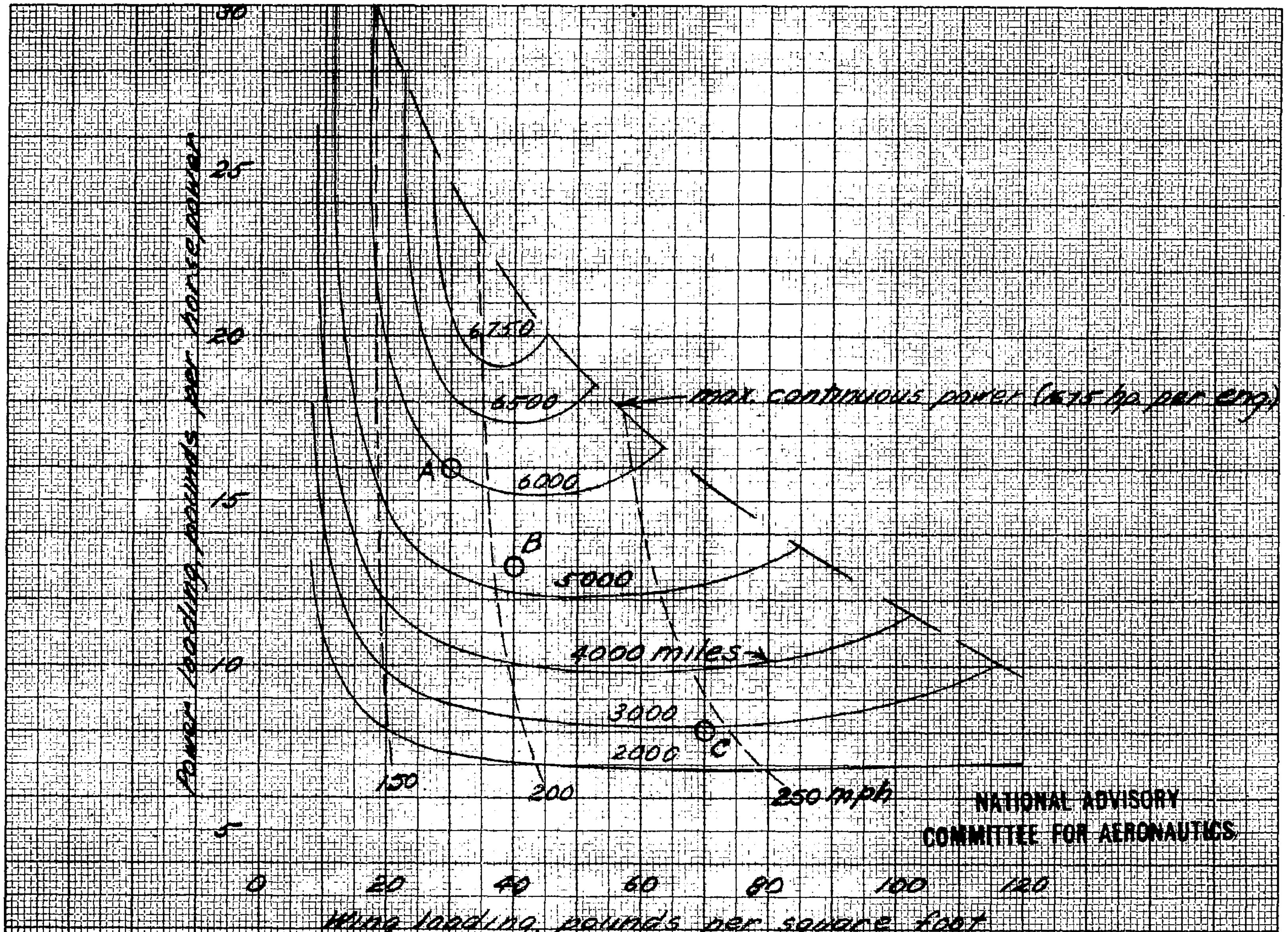





Figure 3(b) - RANGE OF 20000 FOOT DESIGN ALTITUDE AIRPLANES OPERATING AT MAX L/D AND DESIGN ALTITUDE.

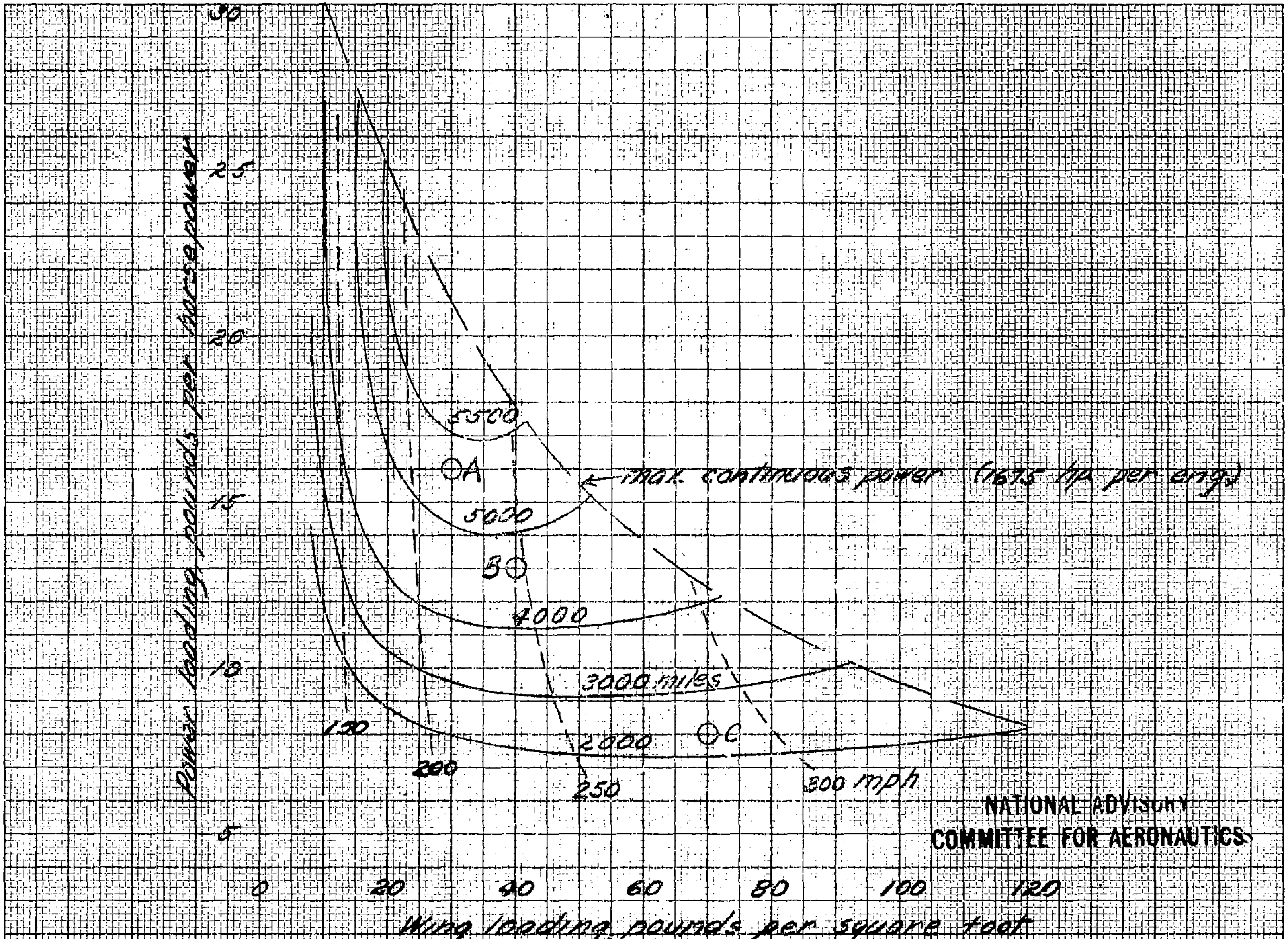



Figure 3(c) - RANGE OF 30000 FOOT DESIGN ALTITUDE AIRPLANES OPERATING AT MAX L/D AND DESIGN ALTITUDE

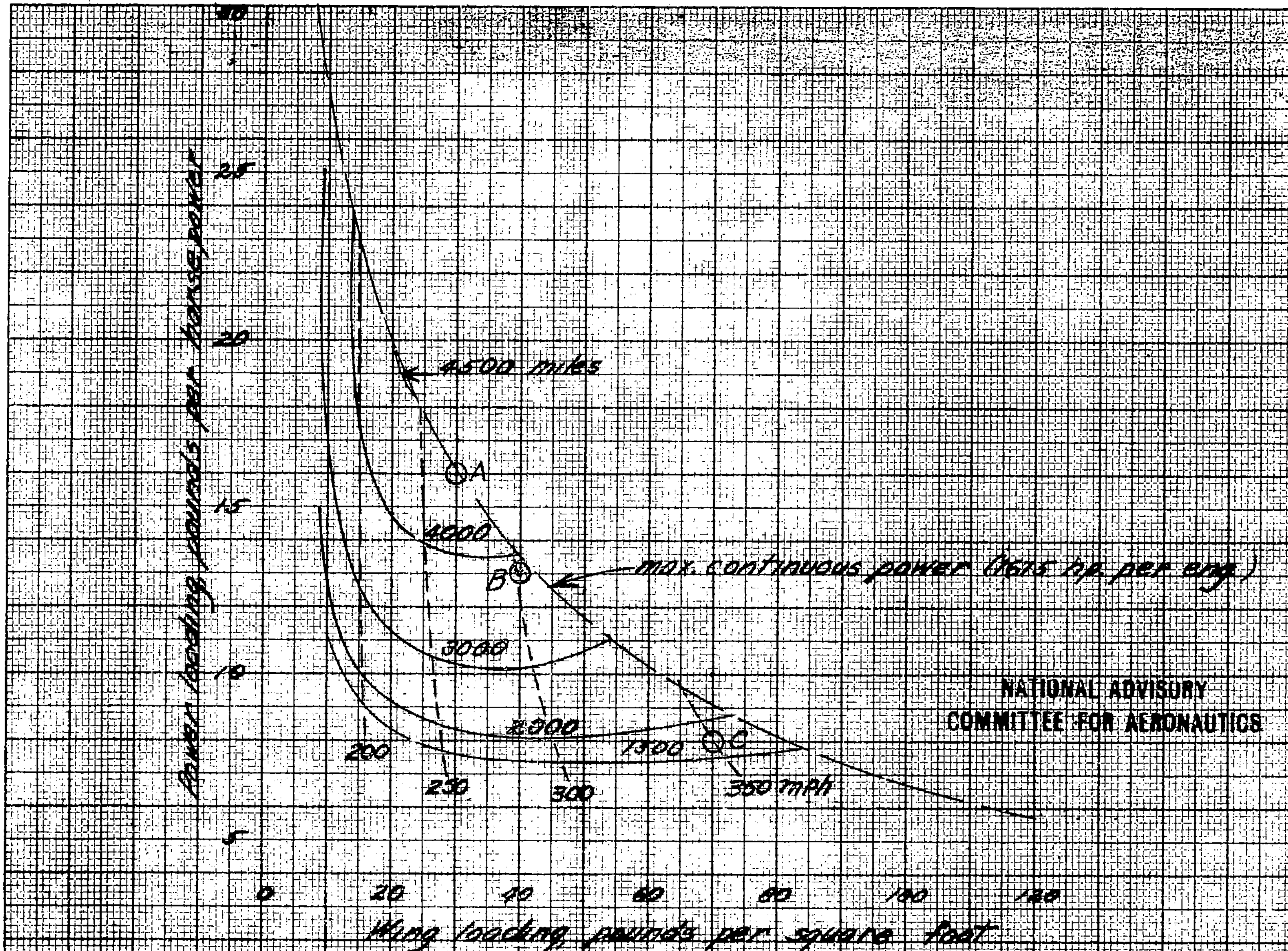



Figure 36(a).- RANGE OF 40,000 FOOT DESIGN ALTITUDE AIRPLANES OPERATING AT MAX L/D AND DESIGN ALTITUDE.

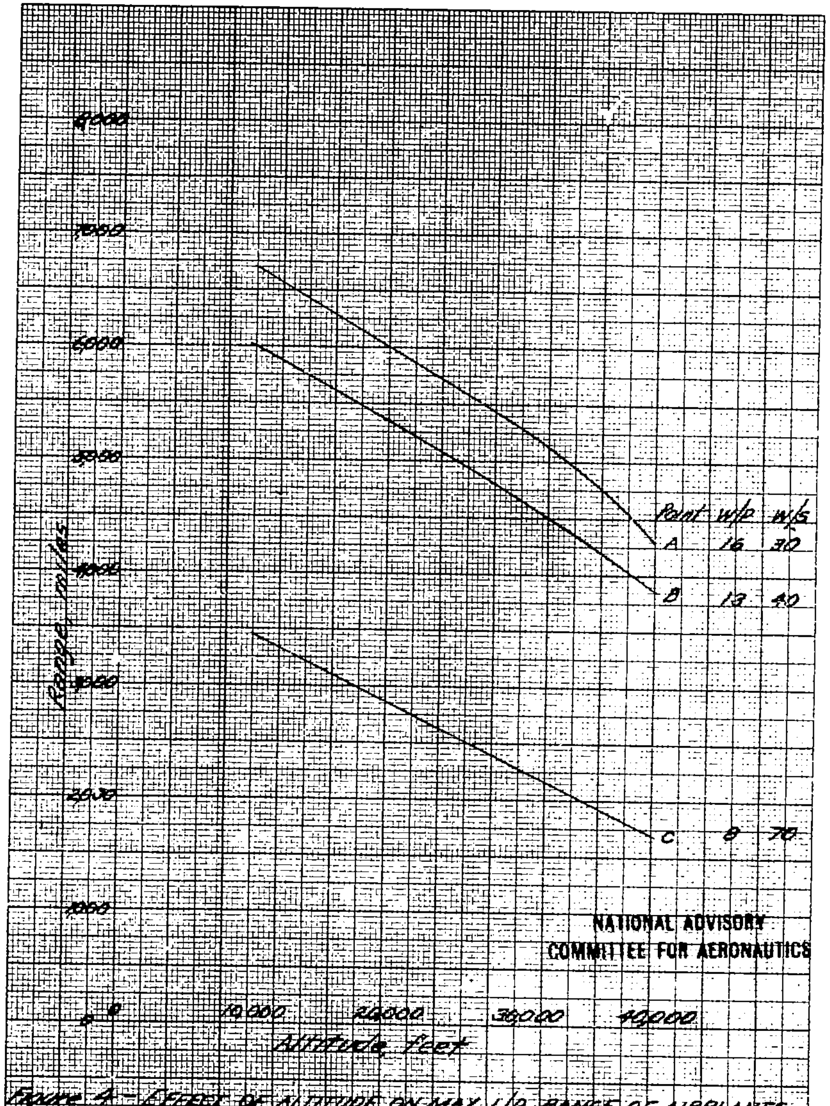



Figure 4 - Effect of altitude on max L/D range of airplanes flying at design altitude, fixed power and wing loading.

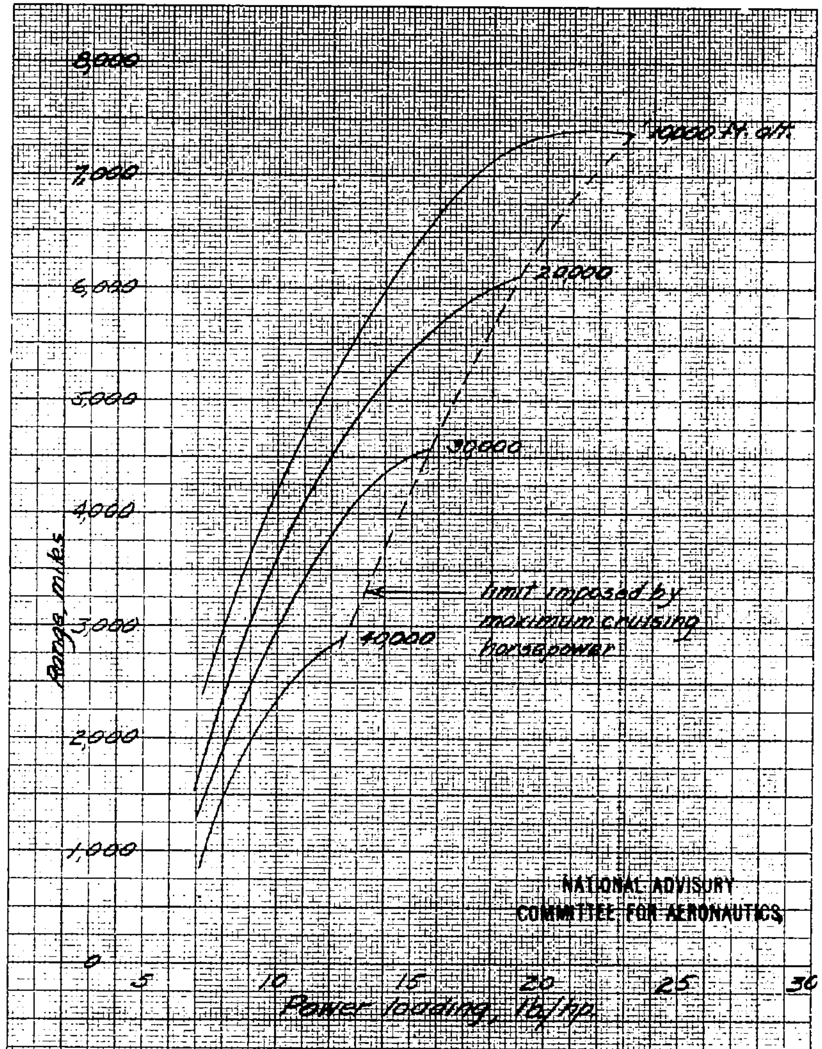



Figure 5 - Effect of design altitude and power loading on range of airplanes with W/S = 50 flying at max L/D at design altitude.

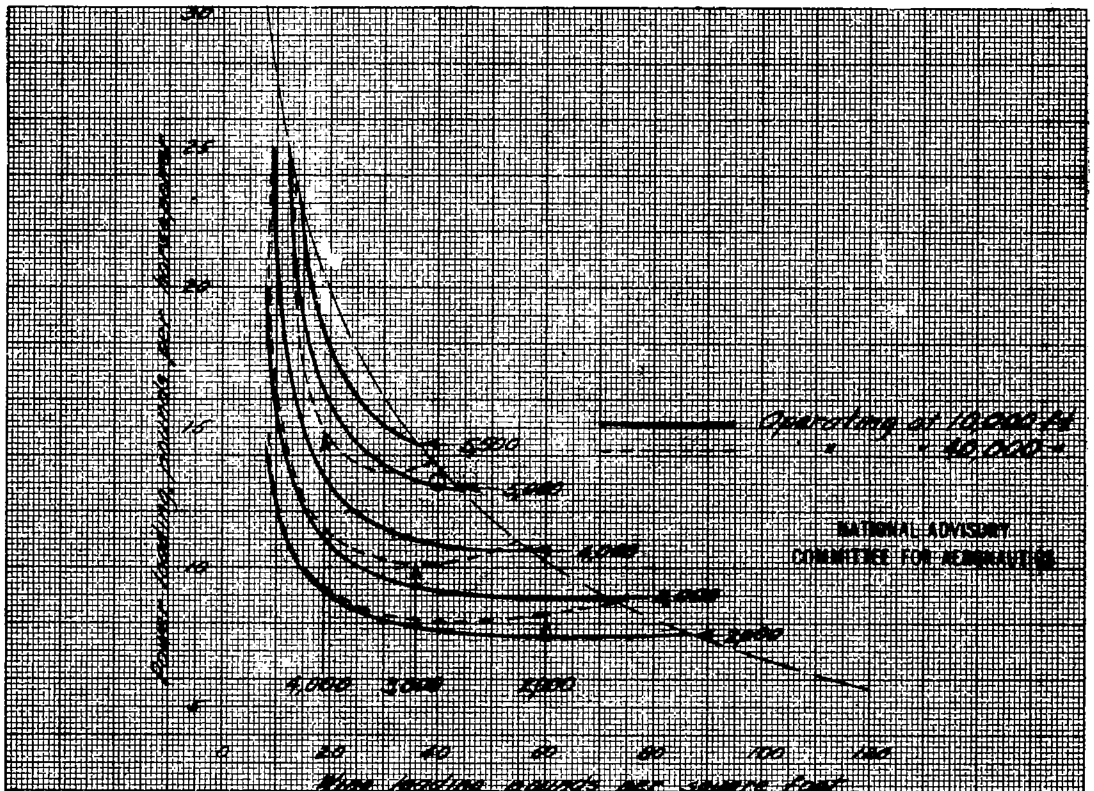



Figure 6.- Range of 40,000 foot design altitude airplanes operating at max l/d at 10,000 feet.

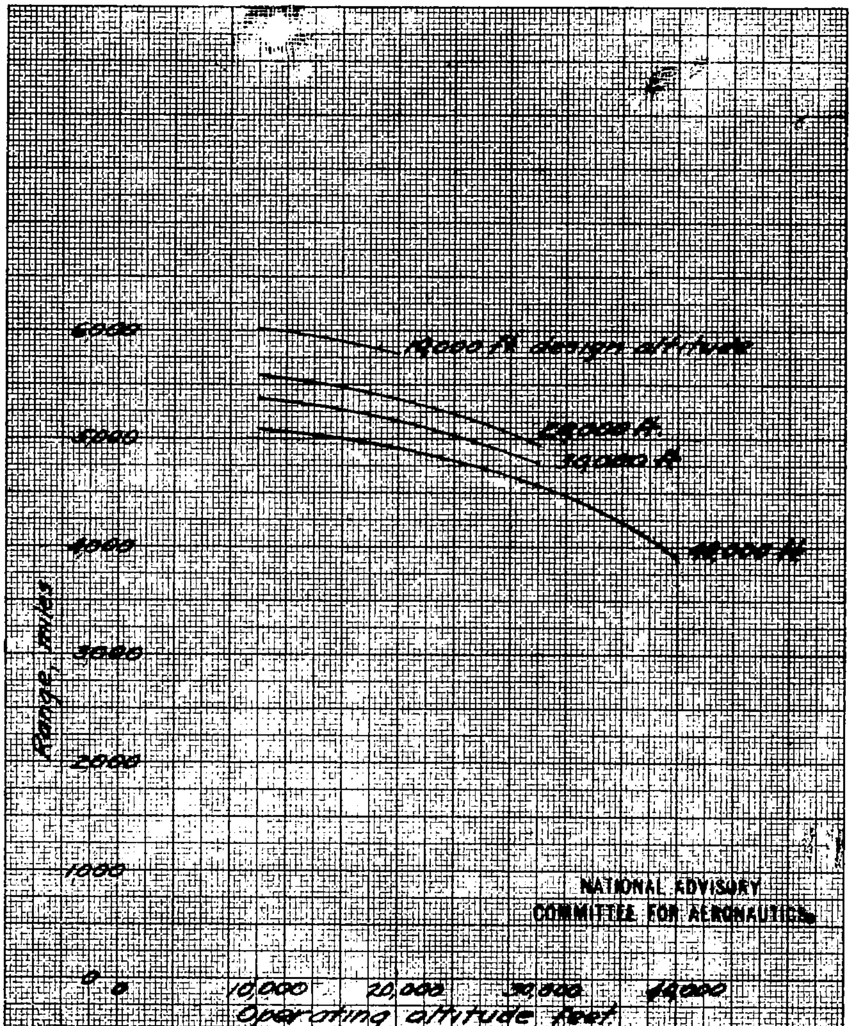



Figure 7 – Range at max. L/D versus operating altitude for airplanes with several design altitudes. [illegible]

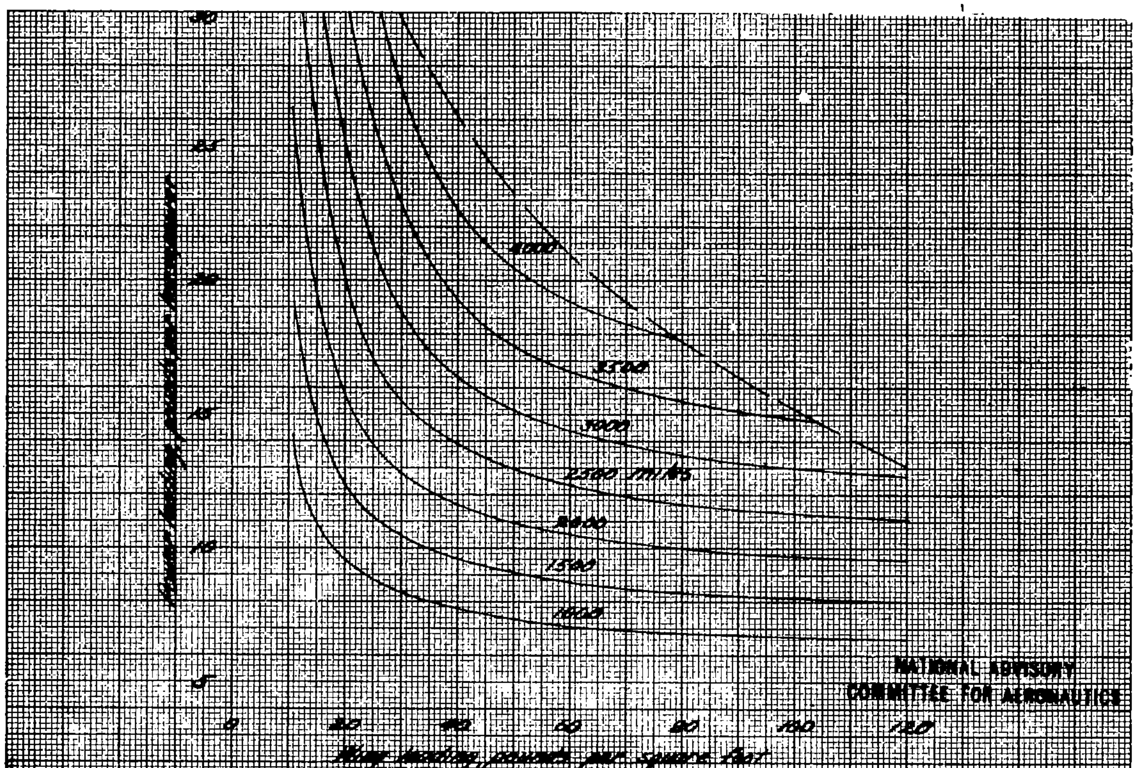



Figure 8(a).- RANGE OF 10000 FOOT DESIGN ALTITUDE AIRPLANES OPERATING AT MAX. CONTINUOUS POWER (1875 HP) AT DESIGN ALTITUDE.

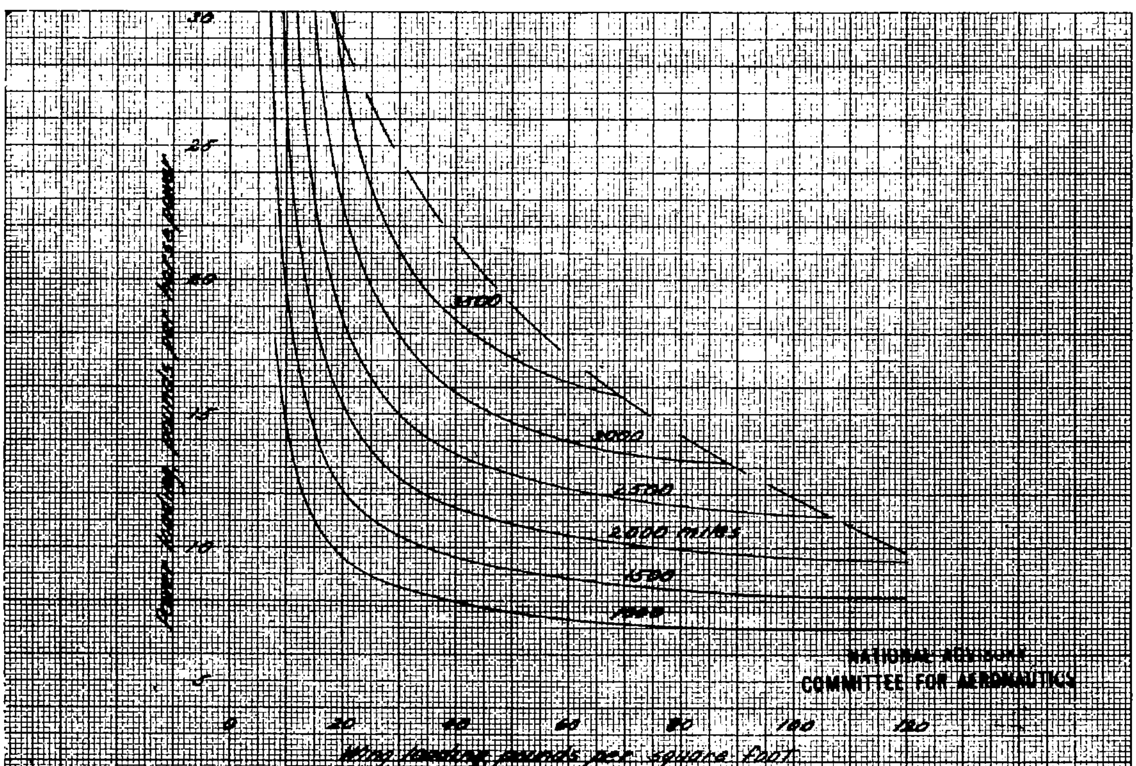



Figure 8(b).- RANGE OF 24000 FOOT DESIGN ALTITUDE AIRPLANES OPERATING AT MAX. CONTINUOUS POWER (1875 HP) AT DESIGN ALTITUDE.

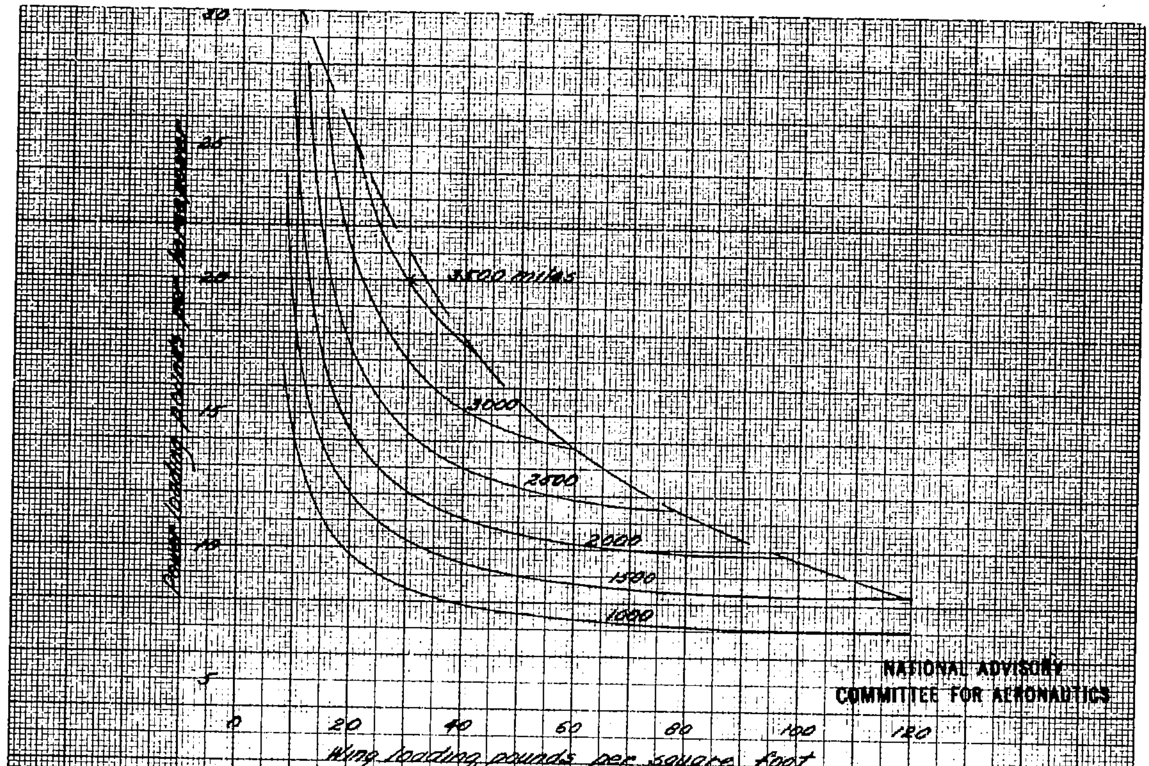



Figure 8(c)- RANGE OF 30000 FOOT DESIGN ALTITUDE AIRPLANES OPERATING AT MAX CONTINUOUS POWER (1675 HP) AT DESIGN ALTITUDE.

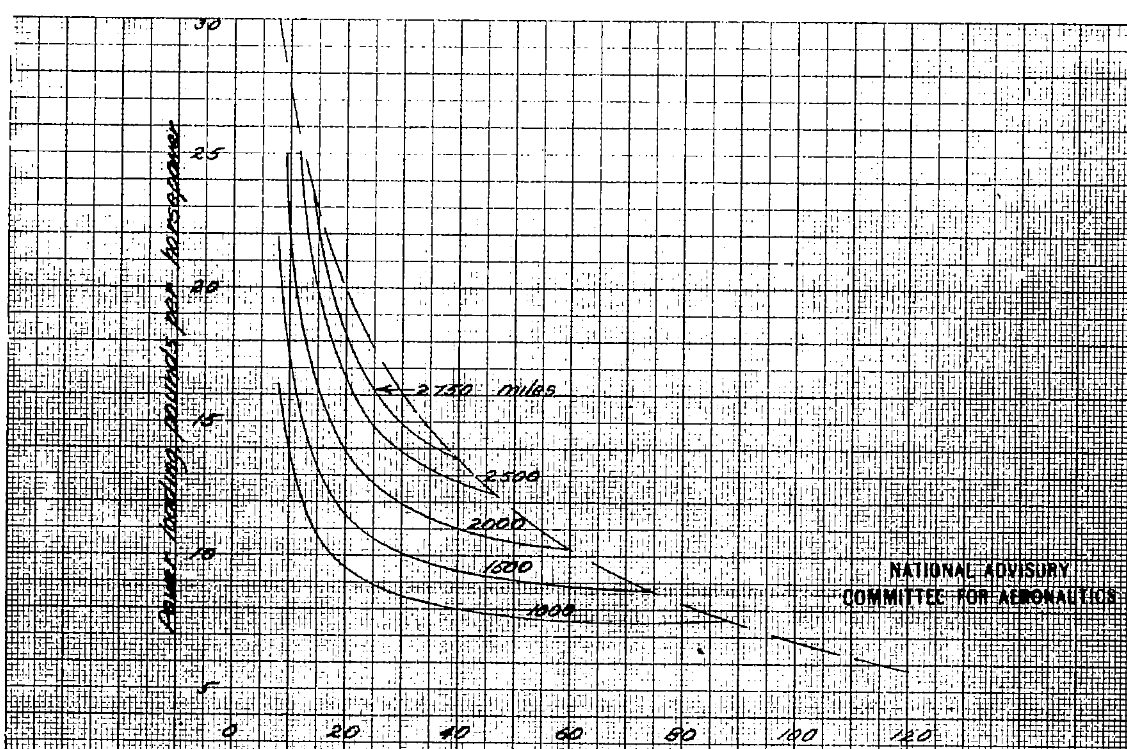



Figure 8(d)- RANGE OF 40000 FOOT DESIGN ALTITUDE AIRPLANES OPERATING AT MAX CONTINUOUS POWER (1675 HP) AT DESIGN ALTITUDE.

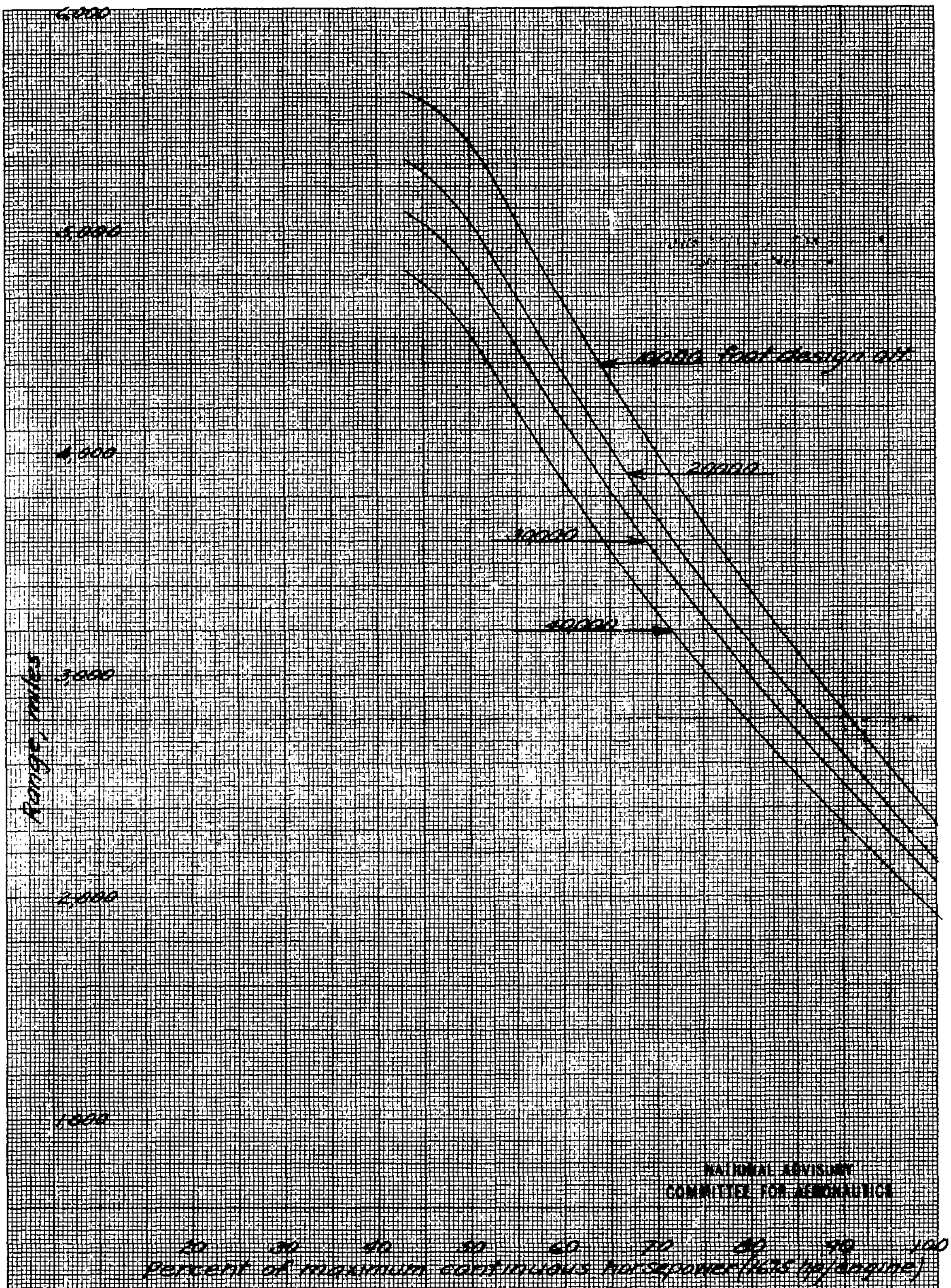



Figure 9a.-EFFECT OF DESIGN ALTITUDE ON RANGE OF AIRPLANES FLYING AT CONSTANT CRUISING POWER AT 10,000 FEET ALTITUDE W/P=15, W/S=40

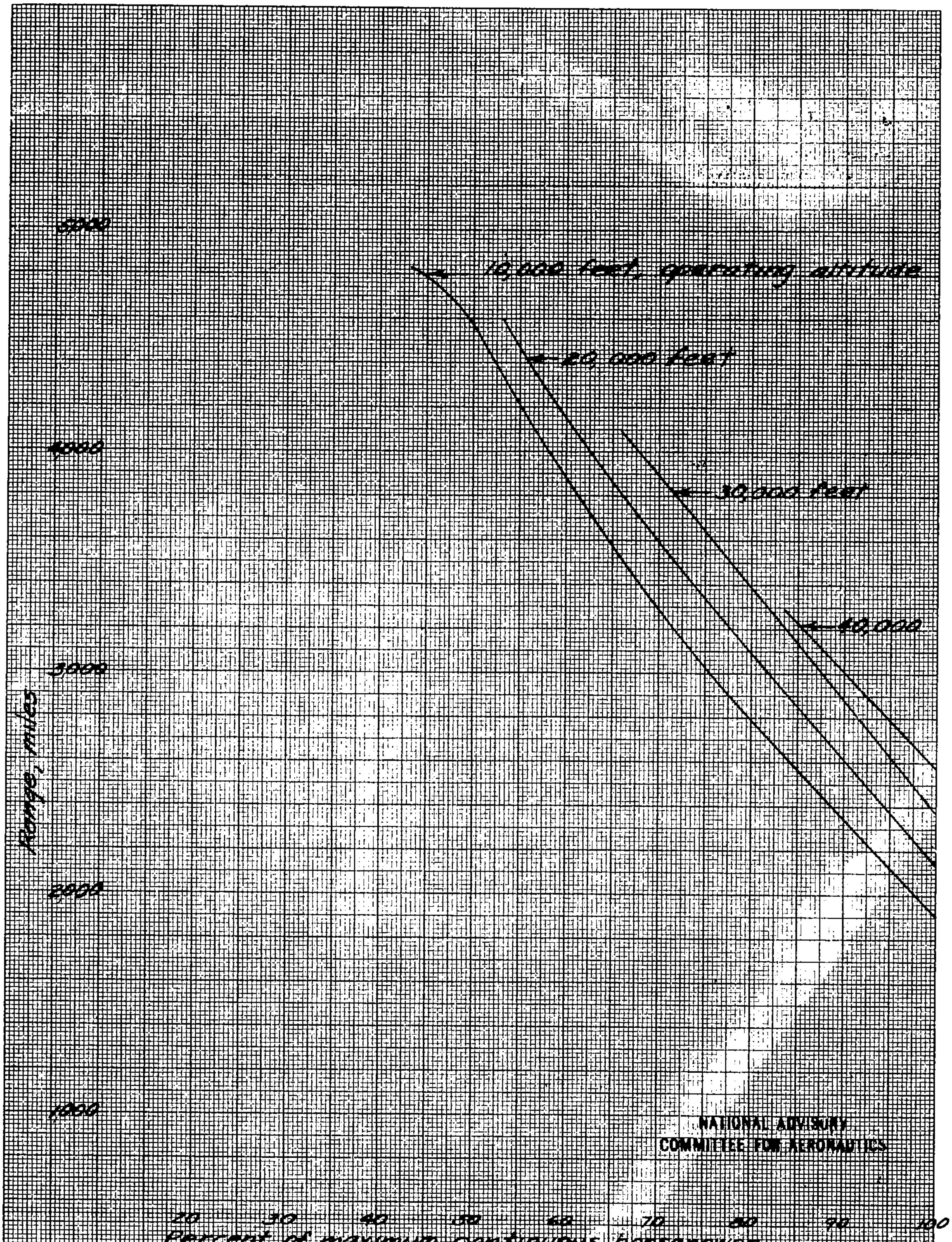



Figure 9(b)- EFFECT OF OPERATING ALTITUDE ON THE RANGE OF A 40,000 FOOT DESIGN ALTITUDE AIRPLANE FLYING AT CONSTANT CRUISING POWER. W/P = 13 W/S = 40.

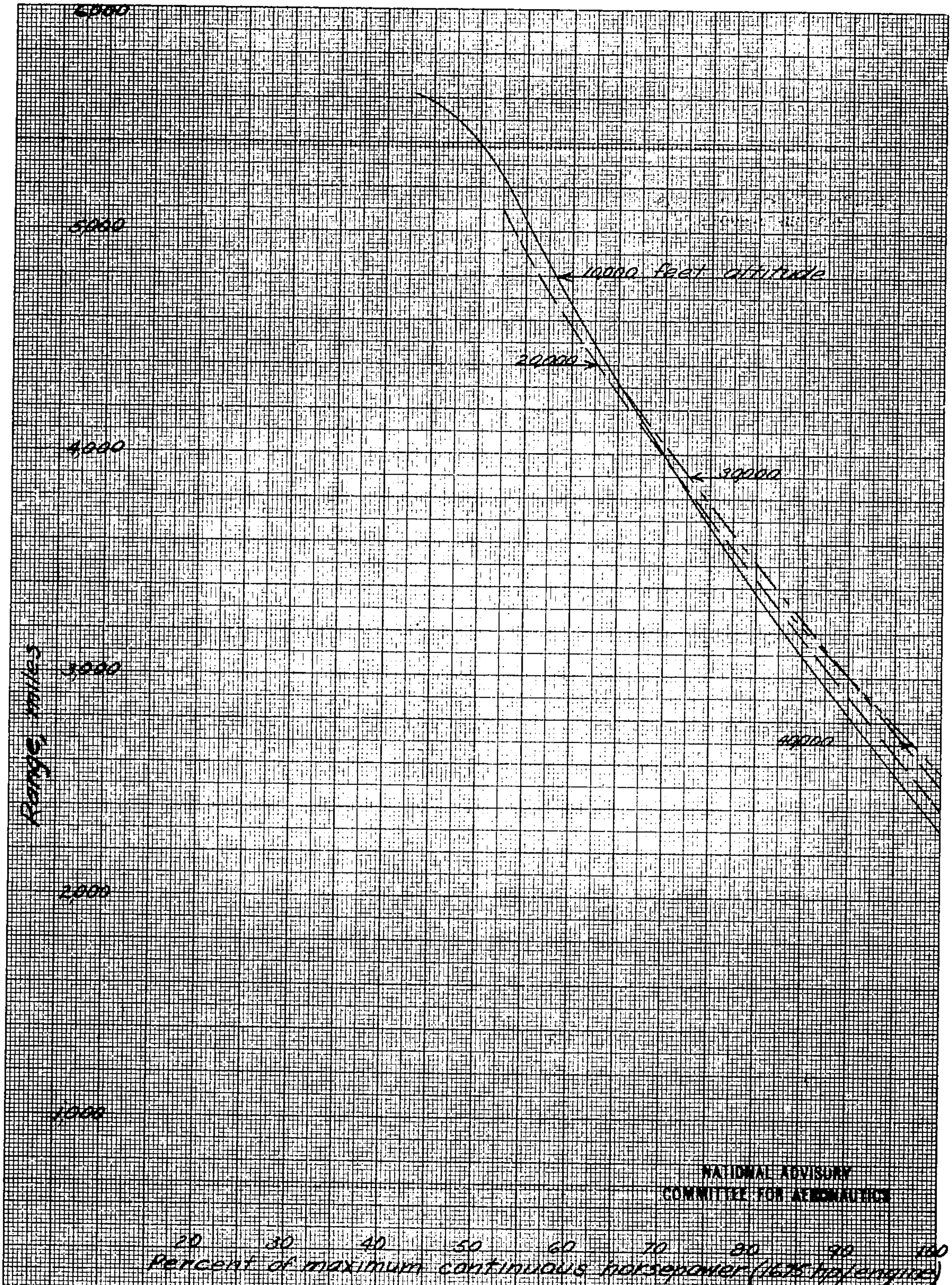



Figure 9(c)- RANGE AT DESIGN ALTITUDE OF AIRPLANES FLYING AT CONSTANT CRUISING POWER. W/P=13 W/P=40.

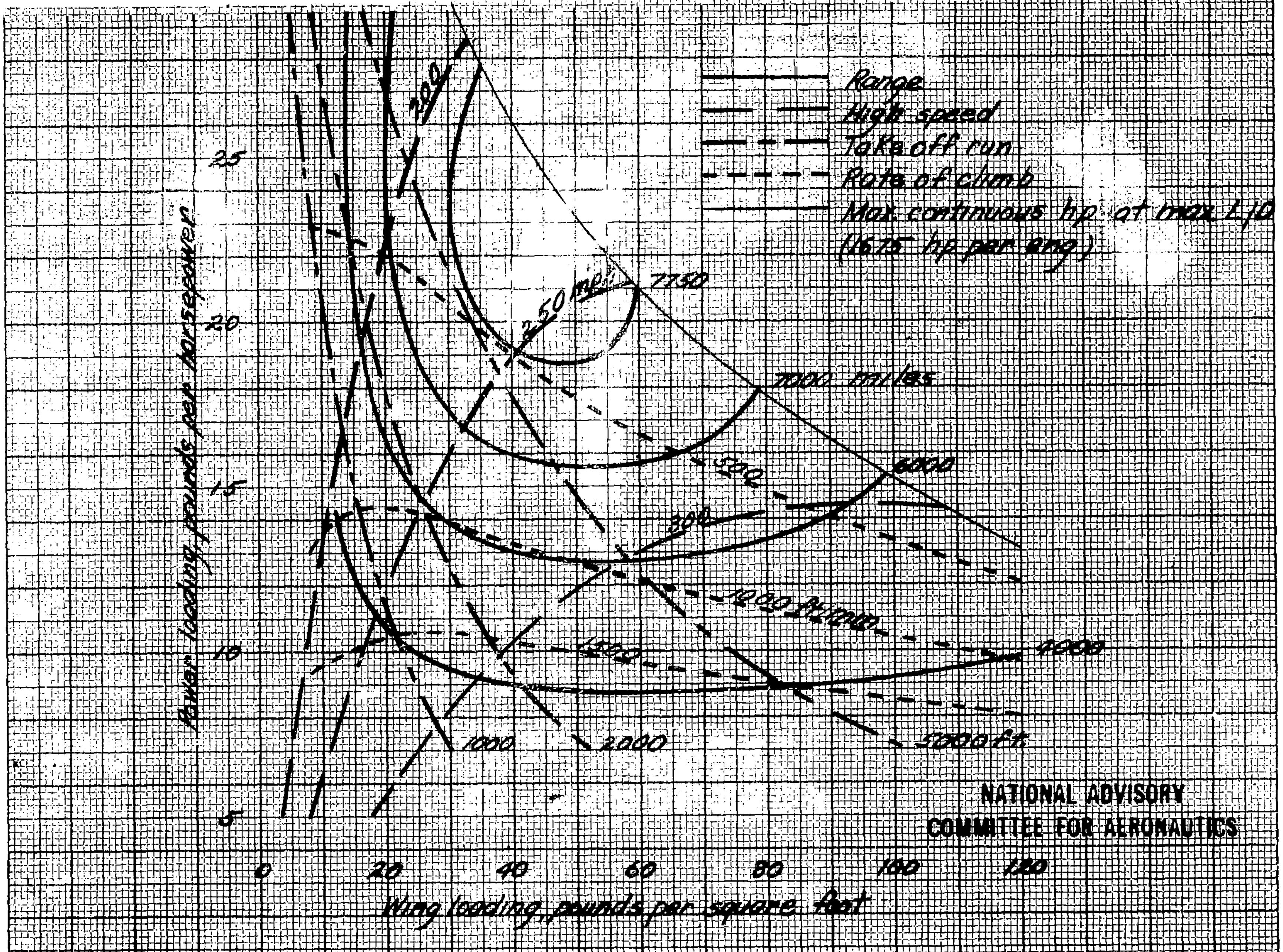
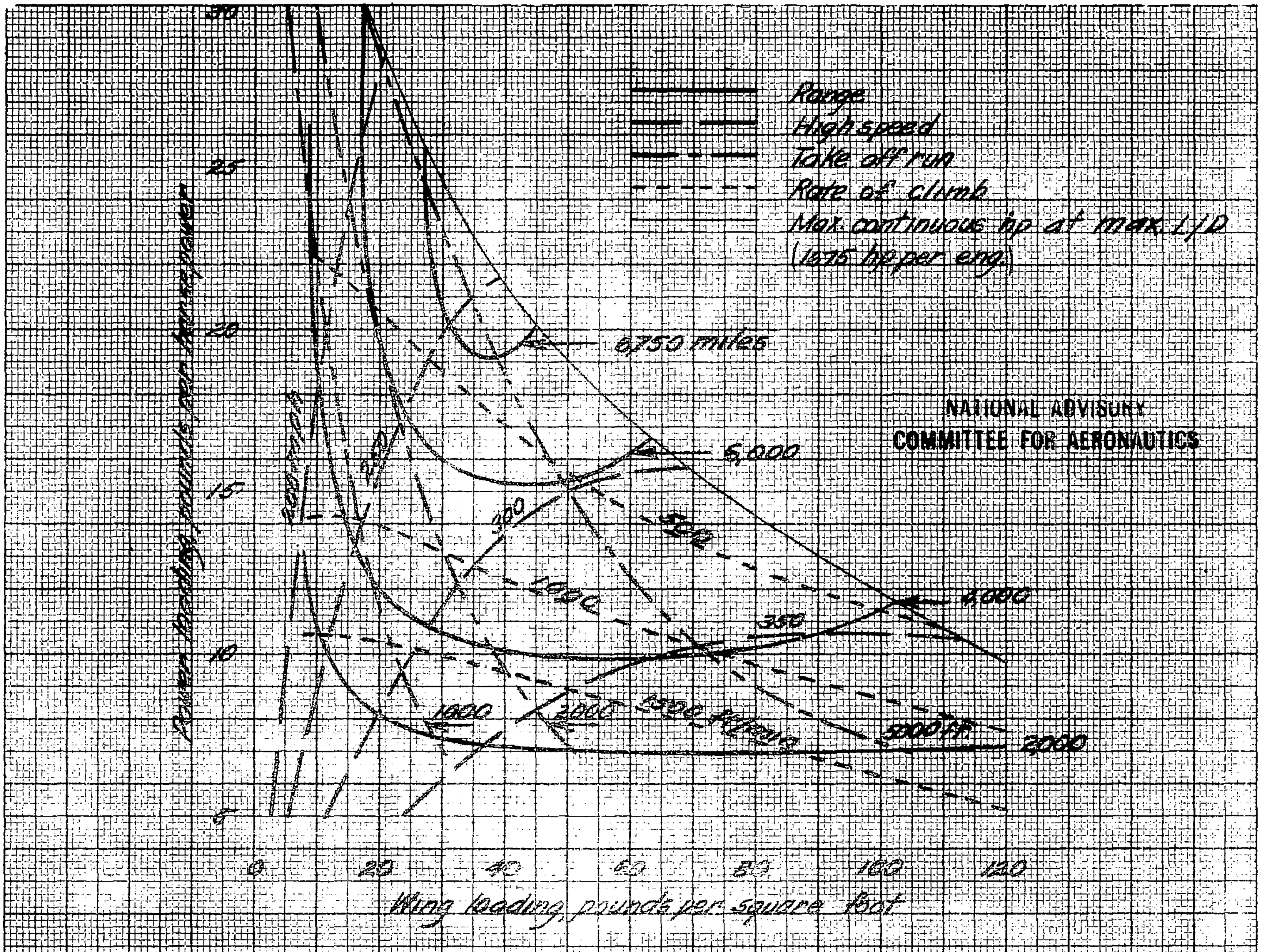



Figure 10(a).- COMPOSITE SELECTION CHART FOR AIRPLANES WITH 10000-FOOT DESIGN ALTITUDE.



NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

Figure 10(b)—COMPOSITE SELECTION CHART FOR AIRPLANES WITH 20000 FOOT DESIGN ALTITUDE.

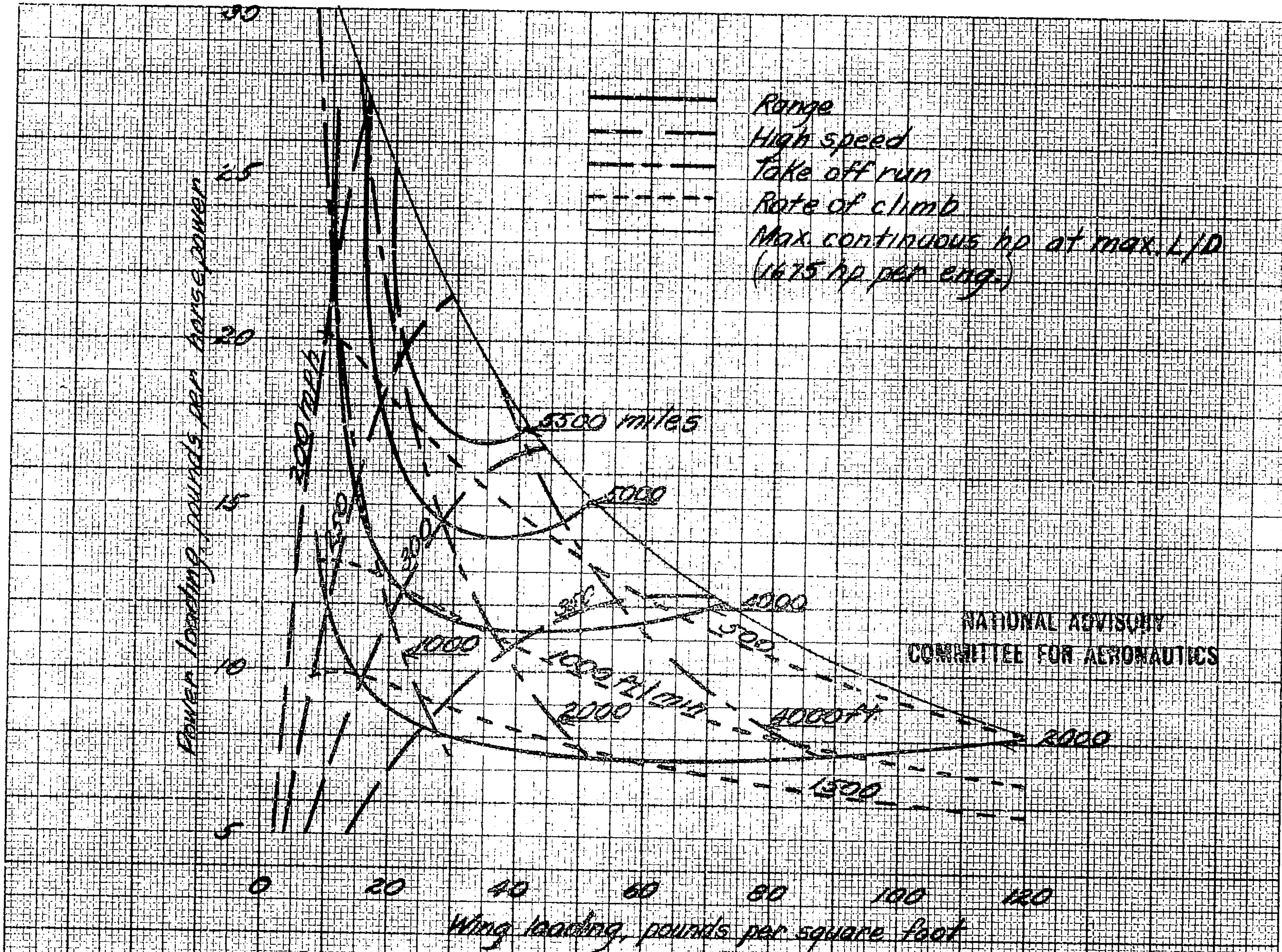





Figure 10(c)-COMPOSITE SELECTION CHART FOR AIRPLANES WITH 30000 FOOT DESIGN ALTITUDE

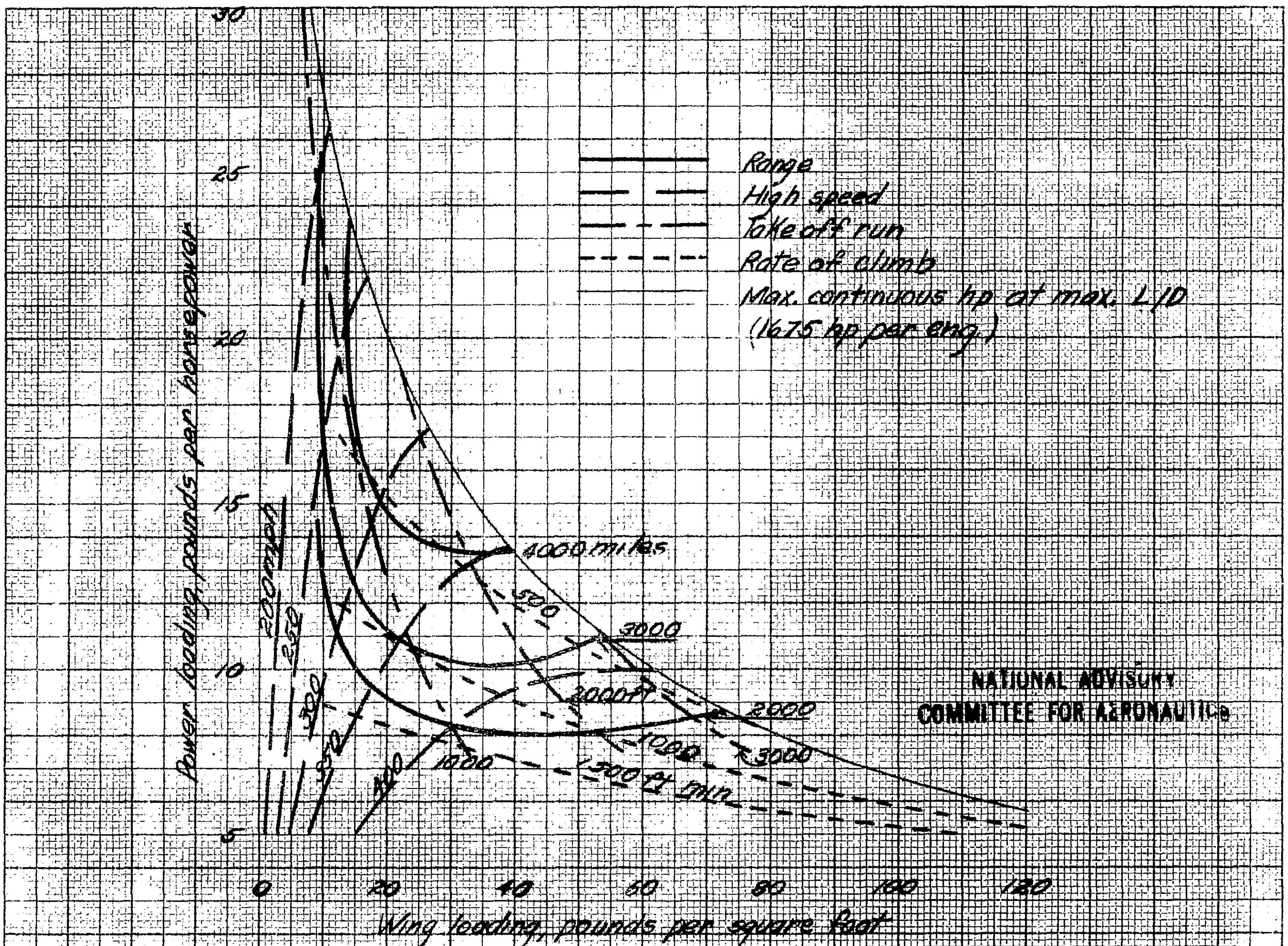



Figure 10(d) - COMPOSITE SELECTION CHART FOR AIRPLANES WITH 10000 FOOT DESIGN ALTITUDE.

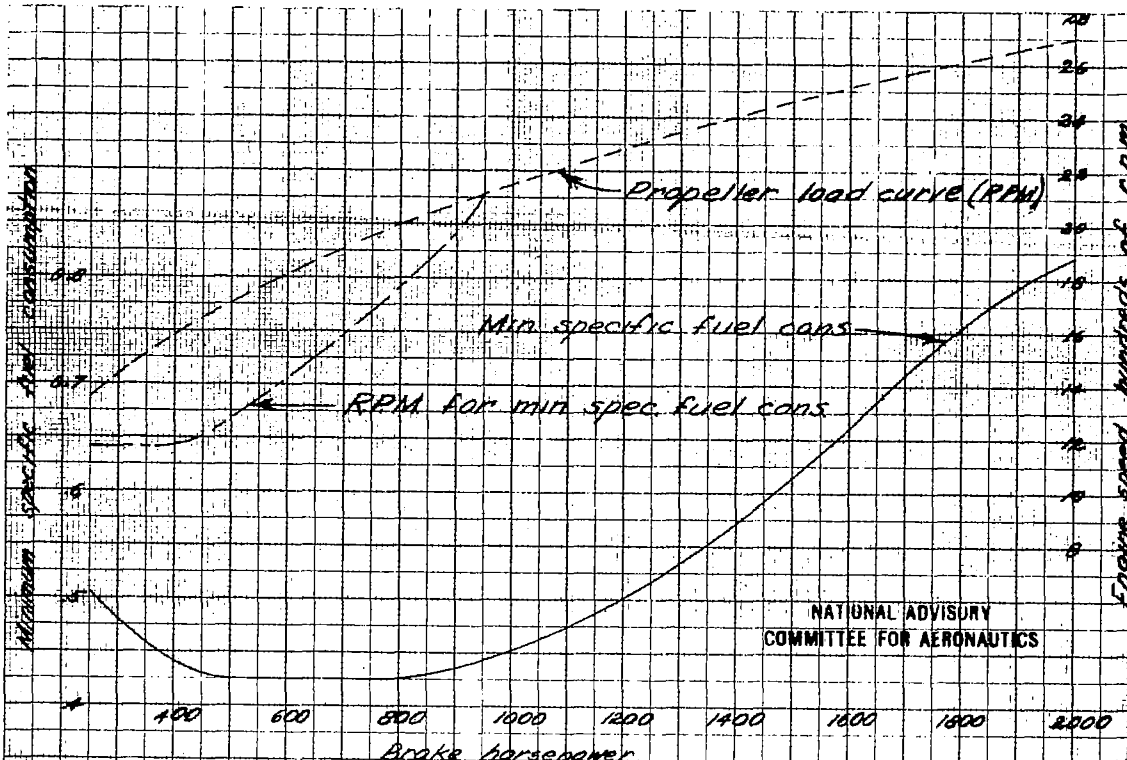



Figure 11(a).- ASSUMED MINIMUM SPECIFIC FUEL CONSUMPTION AND ENGINE R.P.M. FOR MINIMUM SPECIFIC FUEL CONSUMPTION.

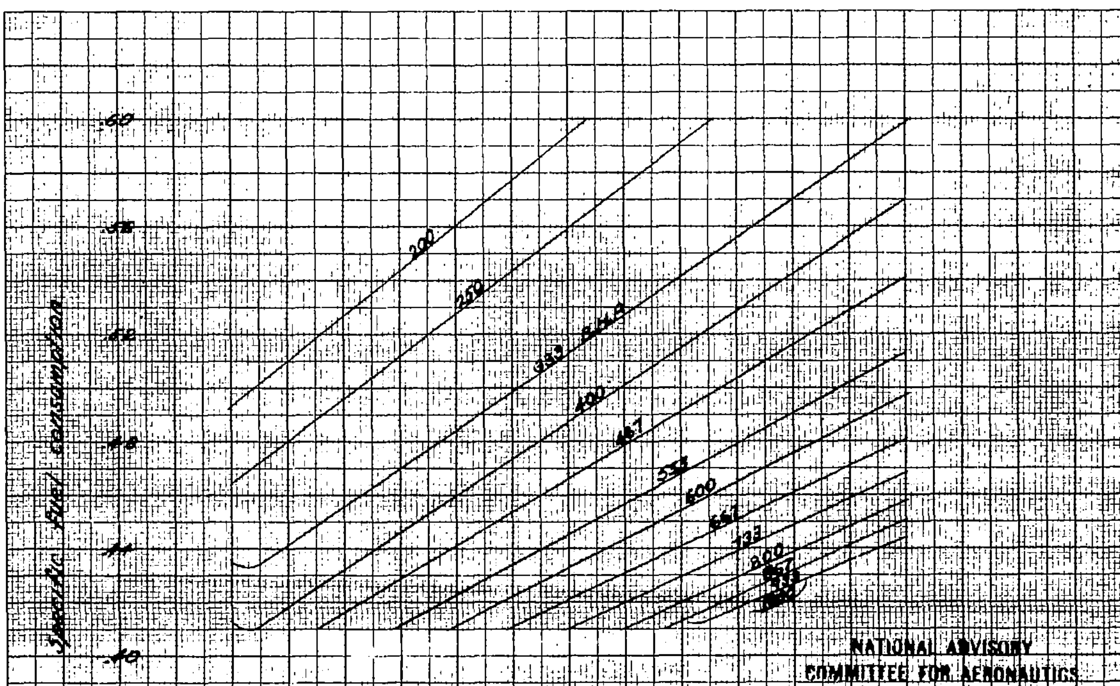



Figure 11(b).- VARIATION OF SPECIFIC FUEL CONSUMPTION WITH ENGINE POWER AND R.P.M.

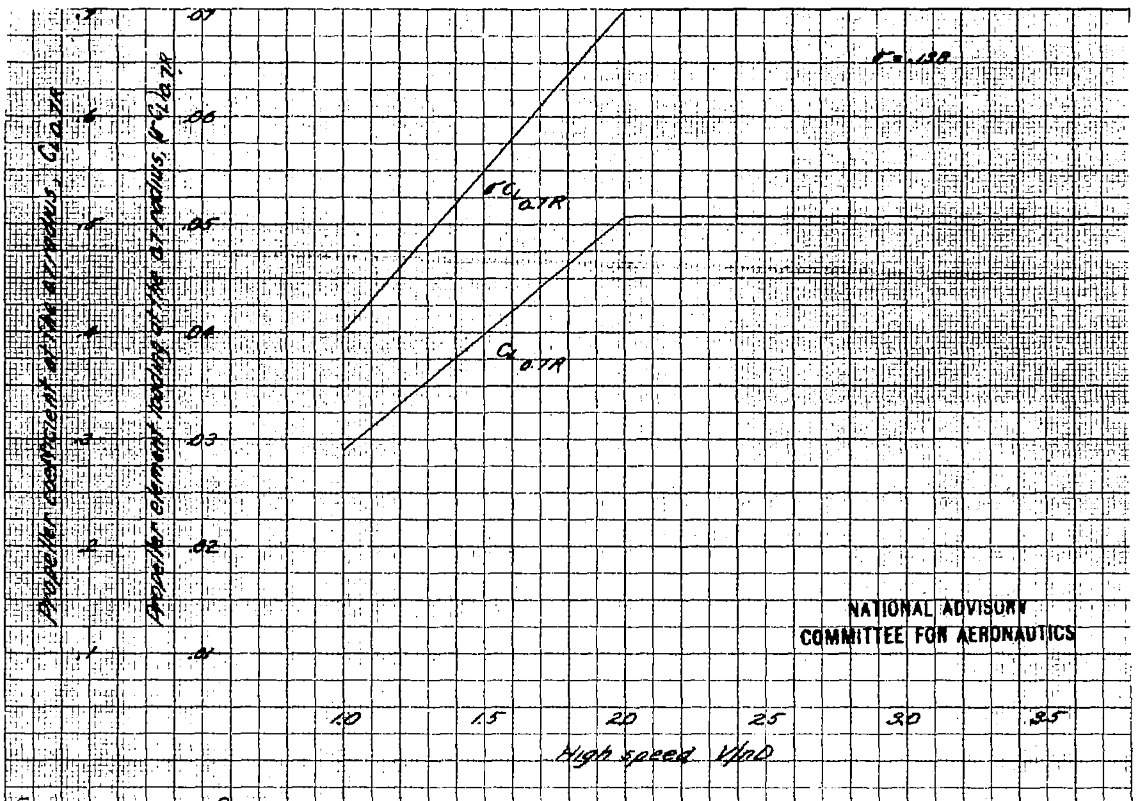



Figure 12 PROPELLER LOADING CRITERIA USED IN PROPELLER SELECTION.

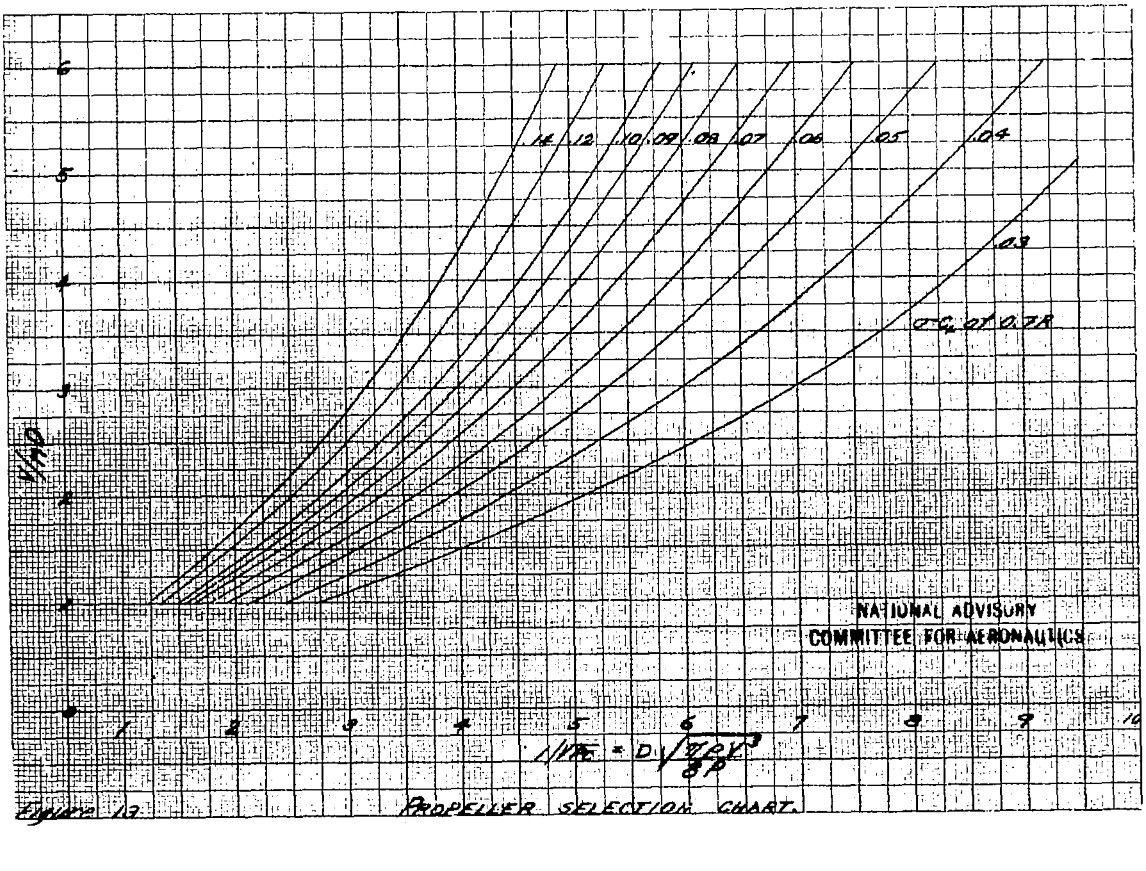



Figure 13 PROPELLER SELECTION CHART.

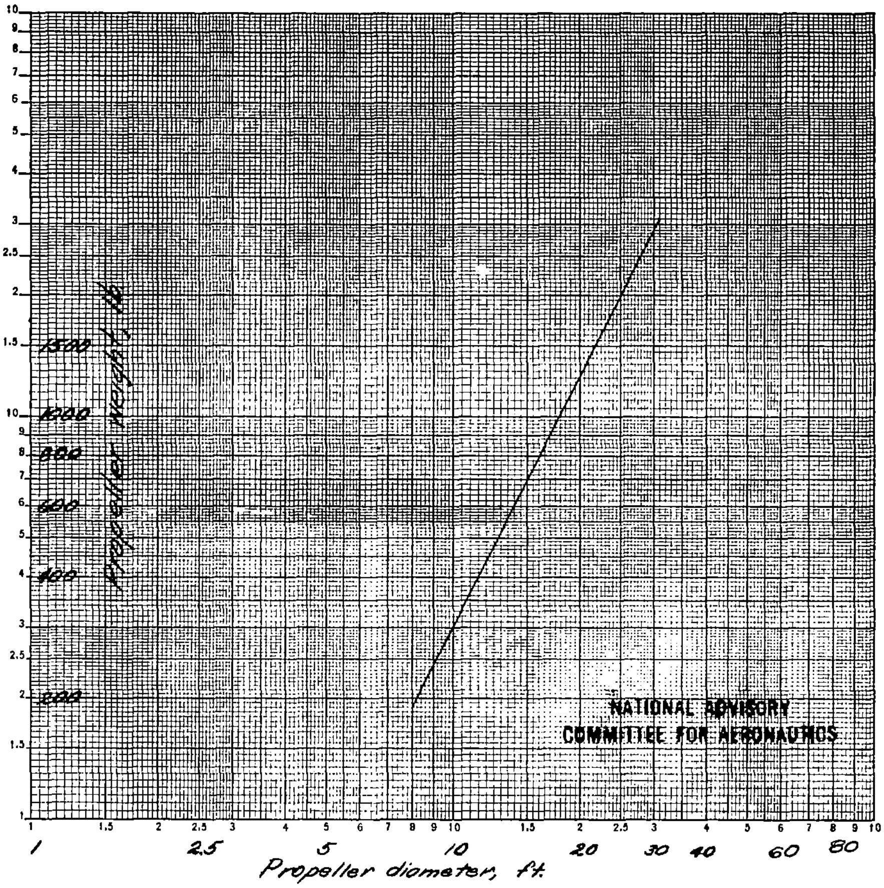



Figure 14. PROPELLER WEIGHT

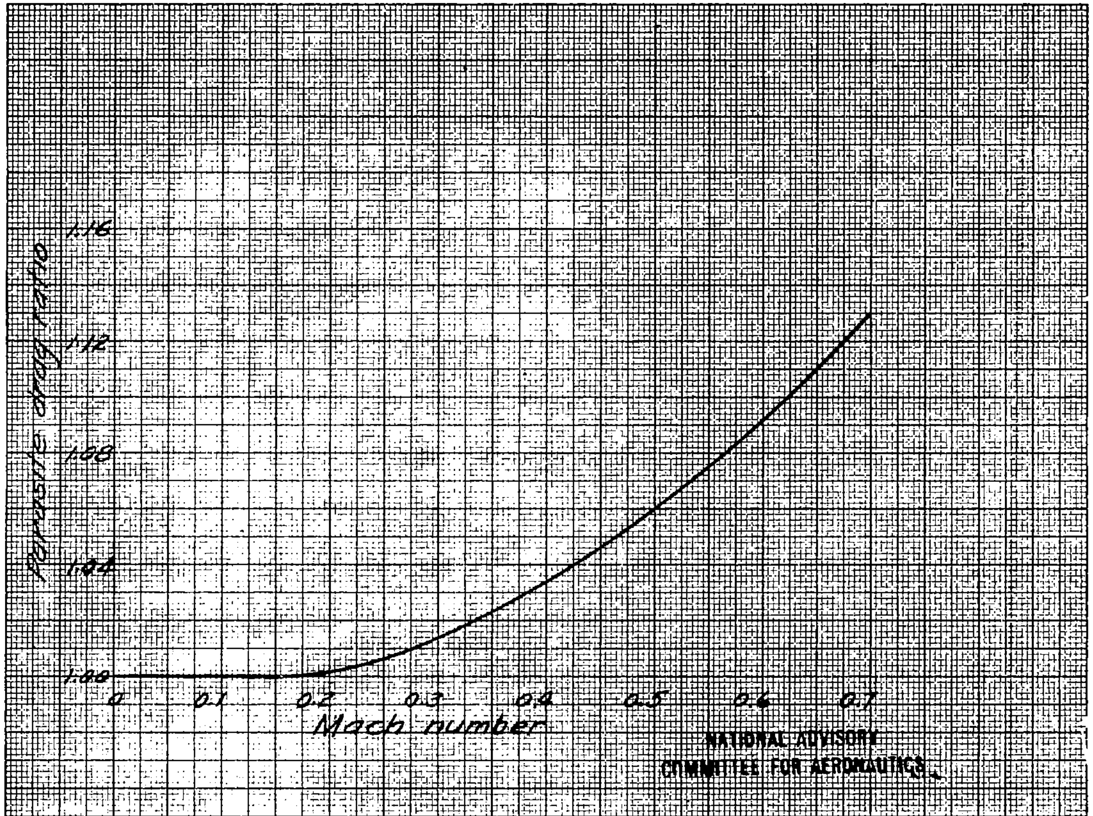



Figure 15.- RATIO OF PARASITE DRAG TO PARASITE DRAG AT LOW SPEED.

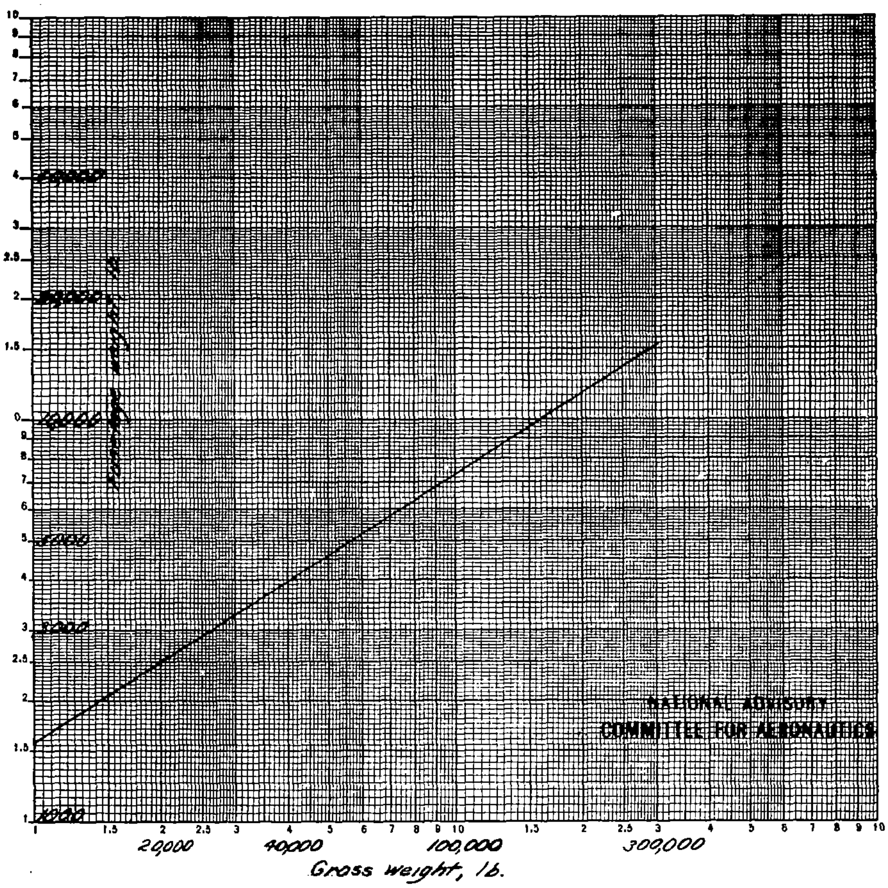



Figure 16. FUSELAGE WEIGHT.

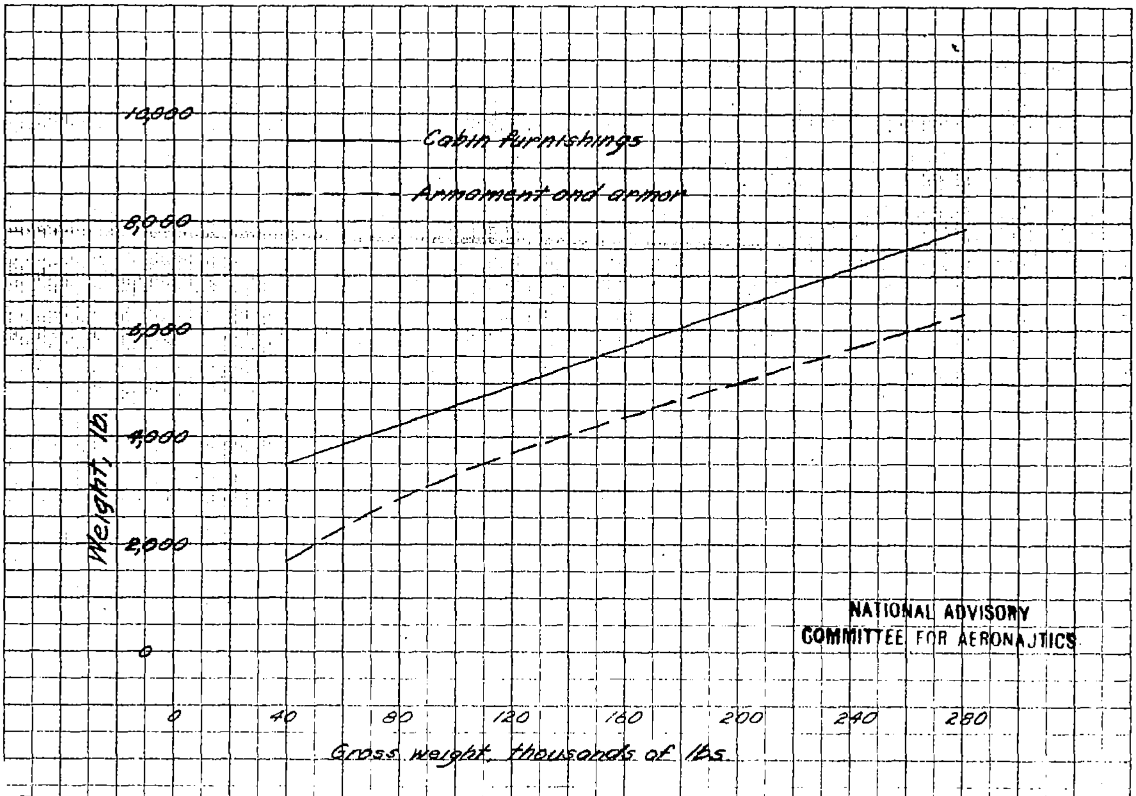



Figure 17(a) - WEIGHT OF CABIN FURNISHINGS, ARMAMENTS AND ARMOR

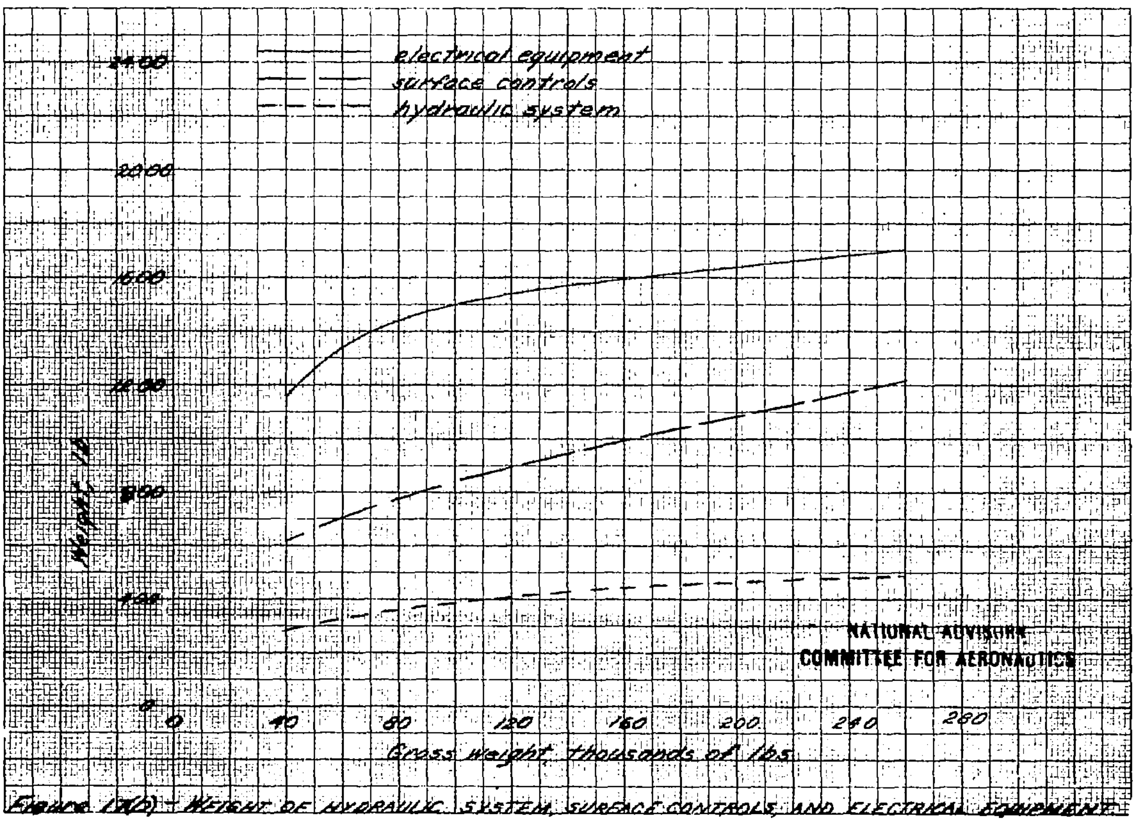



Figure 17(b) - WEIGHT OF HYDRAULIC SYSTEM, SURFACE CONTROLS, AND ELECTRICAL EQUIPMENT

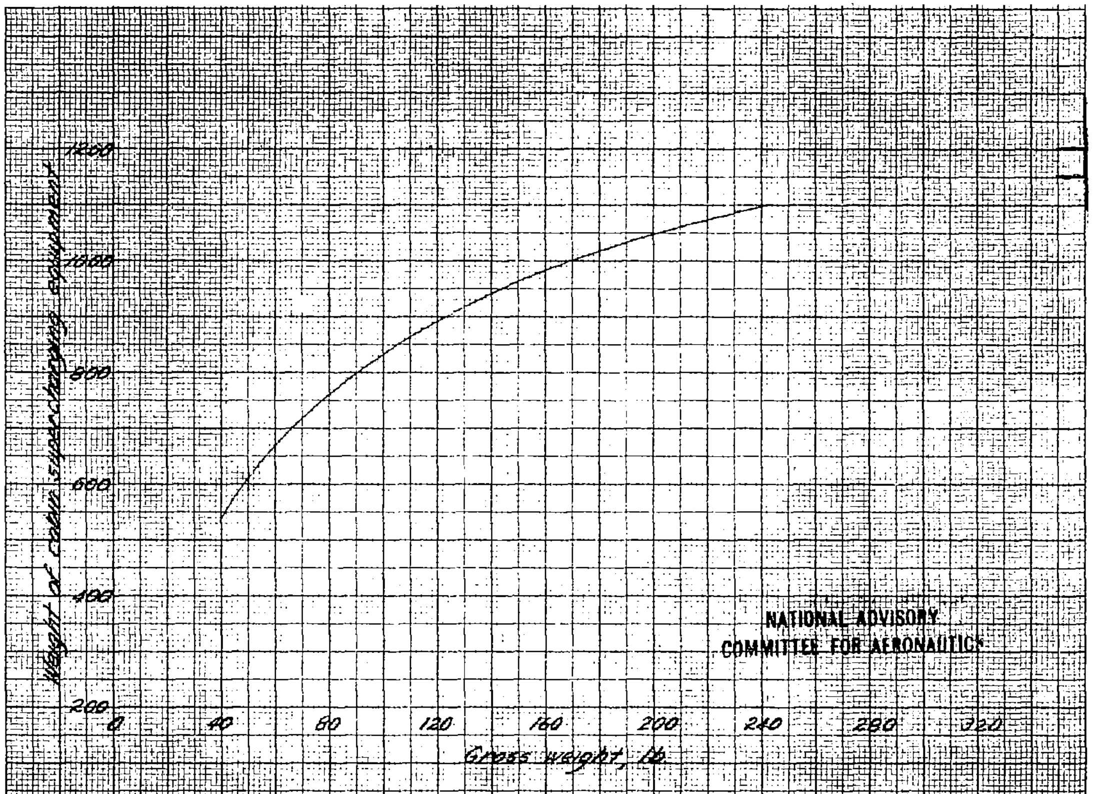



Figure 17(c) - WEIGHT OF CABIN SUPERCHARGING EQUIPMENT FOR 30000 AND 40000 FT. DESIGN ALTITUDES